NASA-CR-172651

LR 29058

CONTRACT NAS1-14000

# Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft



DRL 003

QUARTERLY TECHNICAL REPORT – NO. 13

This report is for the period 1 January 1979 through 31 March 1979

Lockheed Corporation
Lockheed-California Company
Post Office Box 551
Burbank, California 91520



20 April 1979

Prepared for Langley Research Center

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Approved By: Fred English

F.C. English
Program Manager

20 April 1979

Prepared for Langley Research Center

N79-25694#

# FOREWORD

This report was prepared by the Lockheed-California Company, Lockheed Corporation, Burbank, California, under contract NAS1-14000. It is the 13th quarterly technical report, covering work completed between 1 January 1979 and 31 March 1979. The program is sponsored by the National Aeronautics and Space Administration (NASA), Langley Research Center. The program manager for Lockheed is Mr. Fred C. English. Mr. Marvin Dow is project manager for NASA, Langley. The technical representative for NASA, Langley is Mr. Herman L. Bohon.

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SUMMARY

The technical activities performed in this reporting quarter and documented in this report are related to tasks associated with Phase II, Phase III, and Phase IV of the Advanced Composite Vertical Fin (ACVF) Program. These tasks include the following: in Phase II, Component Definition, Material Verification, Process Verification, and Concept Verification; in Phase III, Cover and Spar Fabrication and Test Support; and in Phase IV, Component Tool Development.

Work in process verification and tooling development continued during this reporting period. The decision was made to redesign the ribs to a more producible design. The bead was eliminated and the truss ribs changed to plain C sections. The solid web rib stiffeners were eliminated as well as the beads and the webs are now reinforced with a syntactic core. Syntactic is an epoxy containing glass microballoons.

Two cover specimens were successfully tested in this reporting period. The first specimen (H27) was designed to verify the stability and compression strength of the cover when it is hot and wet. Failure occurred at 120 percent of design ultimate load. As the failure appeared to be fixture induced and was limited to one end of the panel, the remainder of the panel will be retested. The second specimen (H28) was designed to verify the failsafe aspects of the design. The test verified the design.

Preparation of the PRVT test facility is nearing completion and all ten spar durability specimens have been installed in the test chambers. Preparation of the first six cover durability specimens is progressing well. Fabrication, trimming and inspection of the final eight of the twenty cover components is in progress. All the spar static specimens are assembled and awaiting test. The first specimen has been installed in the test fixture. Preparation of cover static specimens for test is just beginning.

A two-thirds section of the full-scale left hand cover was fabricated to verify the tooling. This was considered to be successful and the first full-scale left and right covers were fabricated. Two full-scale rear spars were fabricated during this reporting period.

The indicated weight saving for the ACVF is currently at 27.5 percent (235.6 pounds) including a 10 pound growth allowance. Without the growth allowance a weight saving of 28.6 percent (245.6 pounds) is anticipated. The redesign of the ribs has reduced the anticipated weight saving. Composite material utilization is currently predicted to be 77.4 percent of the redesigned box weight.

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# SECTION 1

# INTRODUCTION

The broad objective of the Aircraft Energy Efficiency (ACEE) Composite Structures Program is to accelerate the use of composite structures in new aircraft by developing technology and processes for early progressive introduction of composite structures into production commercial transport aircraft. This program, as one of several which are collectively aimed toward accomplishing that objective, has a specific objective: to develop and manufacture advanced composite vertical fins for L-1011 transport aircraft. Laboratory tests and analyses will be made to substantiate that the composite fin can be safely and economically operated under service loads and environments and will meet FAA requirements for installation on commercial aircraft. A limited quantity of units will be fabricated to establish manufacturing methods and costs. The Advanced Composite Vertical Fin (ACVF) will make use of advanced composite materials to the maximum extent practical and weigh at least 20 percent less than the metal fin it replaces. A method will be developed to establish cost/weight relationships for the elements of the composite and metal fins to establish cost effective limits for composite applications.

The ACVF to be developed under this program will consist of the entire main box structure of the vertical stabilizer for the L-1011 transport aircraft. The box structure extends from the fuselage production joint to the tip rib and includes the front and rear spars; it is 25 feet tall with a root box chord of 9 feet and represents an area of 150 square feet.

The primary emphasis of this program is to gain a high level of confidence in the structural integrity and durability of advanced composite primary structures. An important secondary objective is to gain sufficient knowledge and experience in manufacturing aircraft structures of advanced composite materials to assess properly its cost-effectiveness.

The duration of this program is 70 months, with completion scheduled November 1982. The master schedule for this program is shown in Figure 1-1. The program is organized in four overlapping phases: Phase II - Design and Analysis; Phase III - Production Readiness Verification Tests (PRVT); Phase IV - Manufacturing Development; and Phase V - Ground Tests and Flight Checkout. Phase I was completed during 1976.

The Lockheed-California Company has teamed with the Lockheed-Georgia Company in the development of the ACVF. Lockheed-California Company, as prime contractor, has overall program responsibility and will design and fabricate the covers and the ribs, conduct the PRVT program, and conduct the full-scale ground tests; Lockheed-Georgia Company will design and fabricate the front, rear, and auxiliary spars, and assemble the composite fin at the plant in Meridian, Mississippi, where the present L-1011 vertical fins are assembled.

Phase I, Engineering Development, has been completed and Phases II, III and IV are in progress.

Phase II, Design and Analysis, consists of completing the detail design and analysis, characterization of the T300/5208 material system, initiating producibility studies, and conducting material, process, and concept verification tests. Phase III, Production Readiness Verification Testing (PRVT) is designed to provide information to answer the following questions:

- What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality levels and quality control procedures?

- What variability in static strength can be expected for production quality components, and are the margins sufficient to account for this variability?

- Will production quality components survive extended-time laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence in in-service durability?

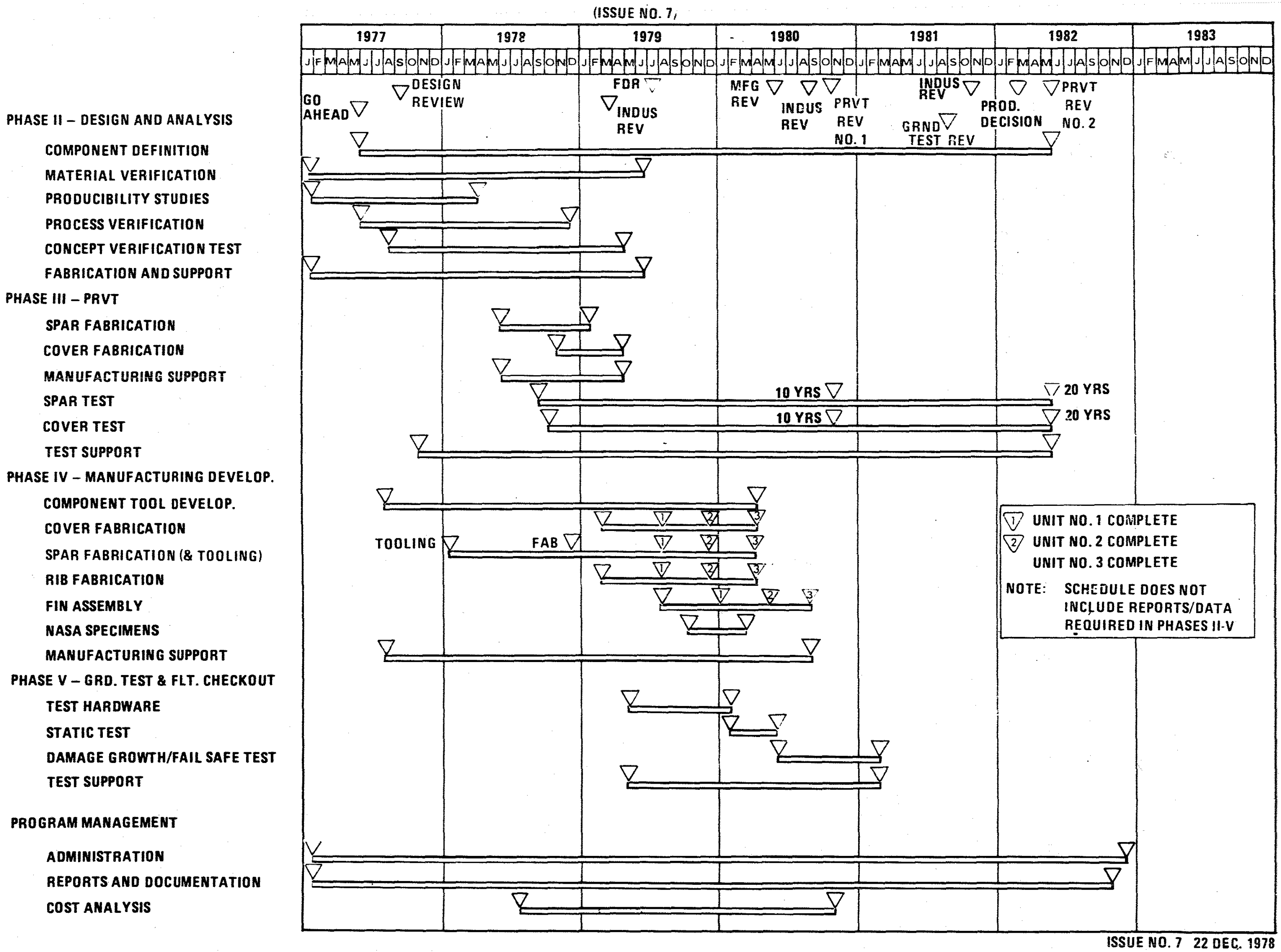



Figure 1-1. ACVF Program Master Schedule

Ten static strength tests and ten durability tests will be conducted on each of two key structural elements of the ACVF. One element will represent the front spar/fuselage attachment area, and the other element will represent the cover/fuselage joint area.

Manufacturing Development, Phase IV, conducted concurrent with Producibility Studies and Process Verification Tasks, will accommodate changes in tooling to take advantage of development of low-cost manufacturing methods. Following NASA's approval of the design, three fins will be fabricated and assembled to prove the design, methods of manufacture, and quality. Actual costs will be documented during fabrication and components will be weighed to update cost and weight estimates.

The manufacturing cost history obtained through the fabrication of the PRVT specimens in a production environment will provide cost data for a starting point for this application of composite structure. Together, they will form the basis for reasonably confident estimates of future production costs.

Ground tests will be conducted on two full-scale fin box beam structures mounted on simulated fuselage support structures during Phase V. The test plan will include static tests, ultimate load and failure load tests on one GTA. Damage growth tests to two lifetimes, and fail-safe and residual strength tests will be done on the second GTA. Repair techniques for in-service maintenance and inspection will be employed throughout tests. Test results will be used to verify the analytical, design, and fabrication procedures, and are essential inputs to the FAA for certification of the aircraft with the ACVF installed. Certification will be based on satisfying both static strength and fail-safe requirements.

Throughout this program, technical information gathered during performance of the contract will be disseminated throughout the aircraft industry and Government. The methods used to distribute this information will be through Quarterly Reports, which will coincide with calendar quarters; and Final Reports of each phase to be distributed at the completion of each phase. All test data and fabrication data will be recorded on Air Force Data Sheets for incorporation in the Air Force Design Guide and Fabrication Guide for Advanced Composites. Oral Reviews will also be conducted at NASA, Langley to acquaint the aircraft industry and the Government with progress of the program.

# SECTION 2

## REVIEW OF QUARTER'S PROGRESS

One of the primary activities involved the process development of the ribs which led to the decision to redesign all the ribs into a more producible form. Several of the new rib drawings have been released and tool fabrication has commenced. Development tools are being used to verify the processing parameters.

The current weight status is shown in Table 2-1. A weight savings of 27.5 percent (235.6 pounds) is currently being predicted including a 10 pound growth allowance. Without the growth allowance, a weight savings of 28.6 percent (245.6 pounds) is anticipated. Composite material use is currently predicted to be 77.4 percent of the redesigned fin box weight. A summary of weight changes since the last quarterly report is presented in Table 2-2. A weight-time history for the composite fin is provided in Figure 2-1.

The material verification testing is continuing. The H12A, "Damage Tolerance", impact testing was completed for both the low speed (tool drop) and the high speed (hail) projectiles. All the panels are currently being ultrasonically inspected. The H13D, "Laminate Durability", tests are progressing well. The fatigue testing will be completed next month. To date no failures have occurred. The H16, "Mechanical Joint", tests are also in progress with fatigue testing two-thirds complete.

The concept verification testing continued with the H27, "Surface Panel Stability", and H28, "Surface Panel Fail Safety", tests being completed. Both these tests are described in detail in Section 4. H27 failed at 120 percent of design ultimate load and H28 exceeded the design requirements. The status of all the concept verification tests is shown in Table 2-3.

## TABLE 2-1. CURRENT WEIGHT STATUS

| Item | Metal Design Total Weight (lb) | Composite Design Target Weight (lb) | Composite Design Total Weight (lb) | Composite Design Composite Mat'l Wt (lb) | Weight Change |
|---|---|---|---|---|---|
| Covers | 460.4 | 368.4 | 351.7 | 333.9 | |
| Spars | 199.0 | 132.0 | 113.8 | 84.5 | -3.4 |
| Ribs | 153.3 | 131.8 | 113.9 | 53.0 | +6.9 |
| Assembly Hardware | 35.4 | 16.7 | 14.6 | - | |
| Protective Finish | 9.6 | 9.6 | 9.6 | - | |
| Lightning Protection | - | 15.5 | 0.0 | - | |
| Installation Penalty | - | 9.0 | 8.5 | - | |
| Design Growth Allowance | - | - | 10.0 | 10.0 | |
| Total Fin Predicted | | | | | |
| Delivery Weight - lb | 857.7 | | 622.1 | 481.4 | +3.5 |
| Weight Saving - lb | | | 235.6 | | |
| Percent Weight Saved | | | 27.5% | | |
| Percent Composite Material | | | | 77.4 | |
| Total Fin Current Ind Indicated Weight - lb (Predicted Less Growth) | | 683.0 | 612.1 28.6% | 471.4 77.0% | |
| Current Indicated Weight of Redesigned Component | 825.4 △1 | | 580.9 | (29.6% Weight Saved) | △2 |

Weight Basis: 5% EST, 81% CALC, 14% ACT

△1 Total metal design weight less weight of components not redesigned

△2 Based on redesigned metal components

TABLE 2-2. SUMMARY OF WEIGHT CHANGES

| Item | Weight Change (lb) | | Remarks |
|---|---|---|---|
| | Total | Composite | |
| Spars | -3.4 | -3.4 | Actual weight of first spar set |
| Ribs | +6.9 | +6.9 | Estimated weight effect of revised rib configuration |
| Total | +3.5 | +3.5 | |

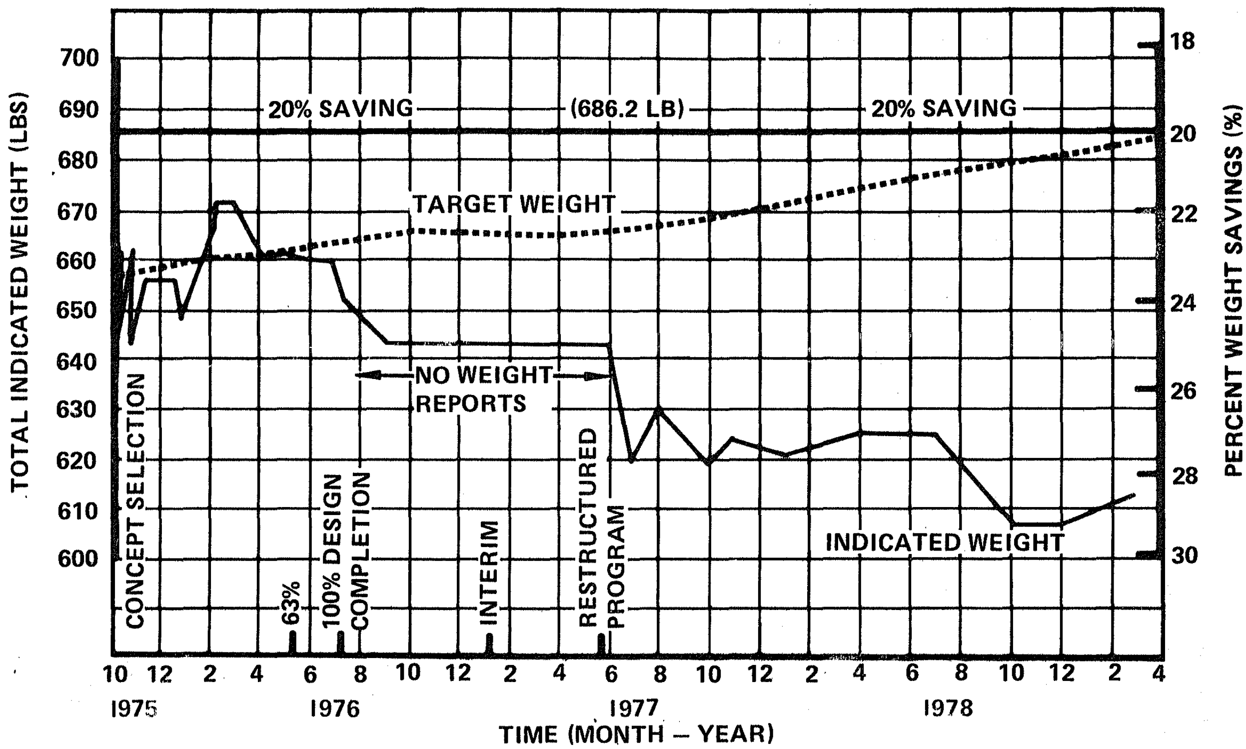



Figure 2-1. Weight-Time History

Preparations for the start of Production Readiness Verification Testing continue. The spar durability specimens were completed and installed in the test chambers. Hook-up is now progressing. The fabrication of cover components continues and twelve have now been delivered to Rye Canyon. Six of these have been designated for durability testing and four are assembled and ready for installation into the test chamber. The first spar static test set-up is complete except for instrumentation.

Fabrication of full-scale spars continued and fabrication of full scale covers commenced. To date one front spar and two rear spars have been completed. A two-thirds section of the full-scale left hand cover was fabricated to prove the tooling and then one full-scale left hand and one full-scale right hand cover were fabricated. Modifications to the fin assembly fixture at Meridian, Mississippi were completed.

TABLE 2-3. STATUS OF CONCEPT VERIFICATION TESTS

| Test No. | Test Description | Type Test/ Condition | Test Status/Results |
|---|---|---|---|
| Covers | | | |
| H-25 | Surface Attach to Fuselage | Static - Compression RT Dry | Test Complete - Failed at 141% of Design Ultimate Load |
| H-26A | Stiffener Runout | Static - Tension Wet - Temp Cycled | Test Specimen Assembly in Progress |
| H-26B | Stiffener Runout | Fatigue - 2 Lifetimes RT Dry | Test Specimen Assembly in Progress |
| H-27 | Surface Panel Stability | Static - Compression - Elevated Temp. Wet | Test Complete Failed at 120% of Design Ultimate Load. See Section 4 |
| H-28 | Surface Panel Fail Safety | Fatigue for 1/2 Lifetimes RT Dry | Test Complete - See Section 4 |
| H-29 | Lightning Strike | RT Wet | Test Specimens Ready to Shipment to LTRI for test |
| Ribs | | | |
| H-24AT | Rudder Hinge Ftg. - Truss Rib | Static - RT Dry | Test Specimens in Process Development Stage |
| H-24AS | Rudder Hinge Ftg. - Solid Web | Static - RT Dry | Test Specimens in Process Development Stage |
| H-24B1 | Actuator Ftg. to Web Attachment | Static - RT Dry | Test Specimens in Process Development Stage |
| H-24B2 | Actuator Ftg. Web Attachment | Fatigue - RT Dry | Test Specimens in Process Development Stage |
| H-24C | VSS 97.19 Rib | Static - RT Dry | Test Specimens in Process Development Stage |
| H-20A | Rib Beam Cap | Static - RT Dry | Test Specimens in Process Development Stage |
| Spars | | | |
| H-20B | Spar Beam | Static - RT Dry | Test Complete - Failed at 163% of Design Ultimate Load |
| H-20B | Spar Beam | Static - Elevated Temp. Wet | Test Complete - Failed at 181% of Design Ultimate Load |
| H-21A1 | Spar Web | Static - RT Dry | Test Complete - Failed at 113% of Design Ultimate Load |
| H-21A2 | Spar Web | Static - Thermo Cycled | Test Complete - Failed at 123% of Design Ultimate Load |
| H-23 | Spar Joint | Static - RT Dry | Test Complete - Failed at 129% of Design Ultimate Load |

# SECTION 3

# RIB DEVELOPMENT

The rib process development continued this quarter. Problems continued with the bead on all the ribs. Various methods were tried to improve the bead configuration but none was totally successful. A decision was made to eliminate the bead and to redesign all the ribs in a simpler, more producible form.

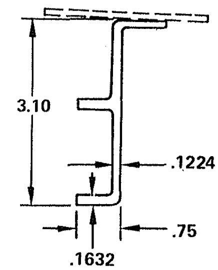
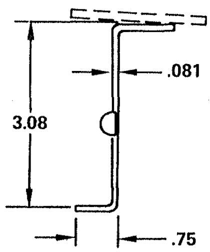
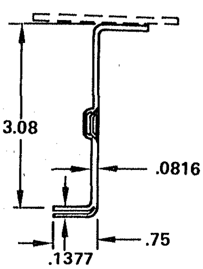
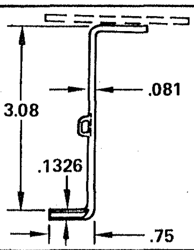
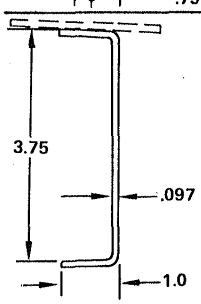
## 3.1 RIB REDESIGN

During this reporting period, the main engineering effort was focused on a variety of rib development problems that have been primarily responsible for delaying construction of full-size rib assemblies and test components. A brief history of the truss rib cap design evolution is shown in Figure 3-1 and described below in chronological order:

Concept A: This is the original rib cap configuration. This design turned out to be highly complex and not cost effective.

Concept B: This concept is the baseline configuration for the comparison shown in Figure 3-1, and represents what was considered a more producible design. In configuration, Concept B is similar to Concept A with the exception of a central bead which replaces the blade stiffener. This concept proved prone to microcracking.

Concept C: In this concept an attempt was made to break up the large amount of $0^{\circ}$ fibers in the bead by interleaving half of the existing $0^{\circ}$ fibers in the outer flange. Although these changes produced a more efficient beam, they also created a loss of bead definition and did not solve the microcracking problem.

| | STRUCTURAL PROPERTIES COMPARISON * | | WEIGHT COMPARISON | PRODUCIBILITY RATING 10=DIFFICULT 1=EASY |
|---|---|---|---|---|
| | AXIAL STIFFNESS | BENDING STIFFNESS | | |
| A   | 1.00 | 1.20 | 1.62 | 10 |
| B   | 1.00 | 1.00 | 1.00 | 3 |
| | BASELINE CONFIGURATION | | | |
| C   | 1.03 | 1.35 | 1.06 | 7 |
| D   | Same as Concept C | Same as Concept C | Same as Concept C | 5 |
| E   | 1.00 | 2.05 | 1.14 | 1 |
| | NEW SELECTED CONFIGURATION | | | |

* Stiffener cutout and effective skin were included

Figure 3-1. Truss Rib Cap Design Evolution

Concept D: This concept is similar to Concept C except for the extra $\pm 45^\circ$ plies added to the central bead to further break-up the concentration of $0^\circ$ plies. This addition produced only slight improvement in over-all bead quality.

The majority of the above problems can be directly traced to the unidirectional bead-stiffener located in the web.

The producibility problems encountered with the bead-stiffener consisted of: (a) bead profile flattening; (b) lateral shifting; and (c) microcracking. These are discussed in more detail later. Some improvement in bead quality was achieved by using a more positive bead constraint combined with other process control refinements. However, in view of the remaining development work to be done on the rib caps, it was decided to halt further manufacturing development on Concept D and concentrate all effort in designing a new set of ribs that would incorporate the following features:

- Low risk/highly producible design.
- Tooling approach compatible with Lockheed experience and process control methods.
- Minimum impact on fixed structural interfaces.
- Proven structural design.
- Low weight.

With the above objectives in mind, a new design concept for the truss rib cap was developed. (See Figure 3-1, Concept E).

Concept E: The shape of the rib cap in Concept E is a basic symmetric channel which will be laid-up and cured on a single male tool. The loss in bending and axial stiffness is compensated for by distributing the $0^\circ$ bead material of Concept D evenly throughout the cross-section and by increasing the depth of the beam section. The small weight penalty experienced by this design is justified due to its greater producibility potential.

It is believed that Concept E will satisfy all design objectives since it does not incorporate the bead stiffener responsible for the difficulties encountered in previous designs.

Solid Web Ribs (Sta. 274.25, 299.97, and 334.25)

The solid web ribs have also undergone several design iterations aimed at making the ribs more producible. Figure 3-2 shows the design evolution of these ribs beginning with the earliest ACVF rib configuration.

Concept A: This was the original solid laminate rib design. The longitudinal web stiffeners were configured as integral blades while the transverse Z stiffeners were secondarily bonded to the web. This design was not cost-effective.

Concept B: This is the baseline and was considered a more producible design. The rib feature $0^{o}$ longitudinal bead stiffeners combined with transverse T shaped stiffeners co-cured to the web.

Concept C: This concept was introduced to correct for several producibility problems encountered in the manufacturer of Concept B. The principal problems concerned profile distortion, lateral shifting, and microcracking within the all $0^{o}$ bead. To correct for these discrepancies, $\pm 45^{o}$ plies were interleaved between the $0^{o}$ plies in the bead. In addition, the "T" stiffeners were replaced with bead stiffeners and relocated on the same side of the web as the longitudinal bead for producibility reasons. Despite some minor improvements obtained as a result of the above changes, none of the process verification rib specimen satisfied design standards.

Concept D: This configuration features a .0375 inch. syntactic core and precludes the need for beads or external stiffeners. It manifests the best structural and producibility characteristics of the four concepts at only a 9% weight penalty (approximately 1 lb. per fin). This concept represents the new configuration for the solid web ribs. The syntactic core can be replaced by solid graphite at the cost of only 1.5 lbs per fin. This may represent a cost saving.

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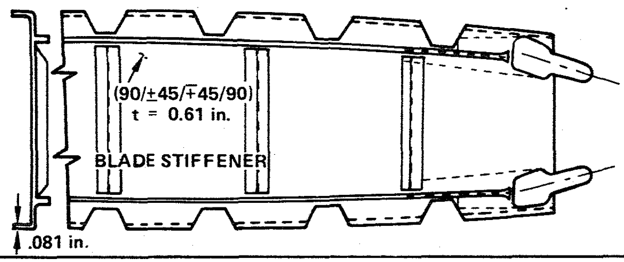
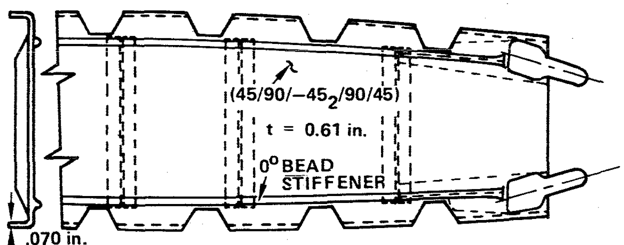
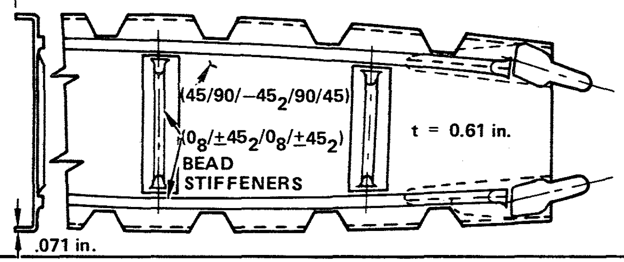
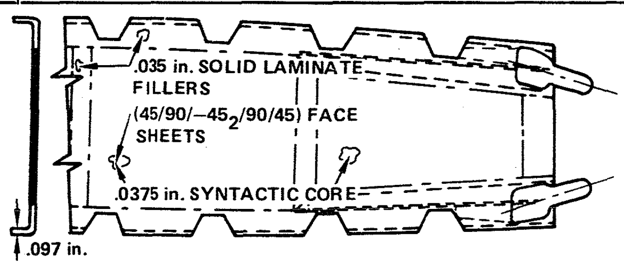
| DESIGN CONCEPT | | STRUCTURAL PROPERTIES COMPARISON | | WEIGHT COMPARISON | PRODUCIBILITY RATING 10=DIFFICULT 1=EASY |
|---|---|---|---|---|---|
| | | TORSIONAL STIFFNESS | WEB BUCKLING RESISTANCE | | |
| A |   | 1.00 | 1.00 | 1.10 | 10 |
| B |   | 1.00 | 1.00 BASELINE CONFIGURATION | 1.00 | 5 |
| C |   | 1.00 | 0.95 | 1.14 | 3 |
| D |   | 1.10 | 2.43 NEW SELECTED CONFIGURATION | 1.09 | 1 |

Figure 3-2. Solid Web Rib Design Evolution

## 3.2 PROCESS AND TOOL DEVELOPMENT

Rib development continued through January 1979 to focus on methods of making stiffening beads which were free of microcracks. The bead design upon which development effort had been concentrated is the "baseline" design which is a mass of all $0^{\circ}$ fibers embedded centrally in the laminate. Various tooling and fabrication approaches were pursued in the attempt to provide a satisfactory bead. However, the best manufacturing effort, which is shown in Figure 3-3, still revealed cracking through the $0^{\circ}$ fibers.

Recognizing that the tendency toward cracking increased with the volume of $0^{\circ}$ fibers in the bead, Engineering proposed alternate concepts for bead design. These concepts provided for moving some of the $0^{\circ}$ fibers from the bead to the flange, and breaking up the amount of these fibers in any single grouping. See Design Concept "C" in Figure 3-1.

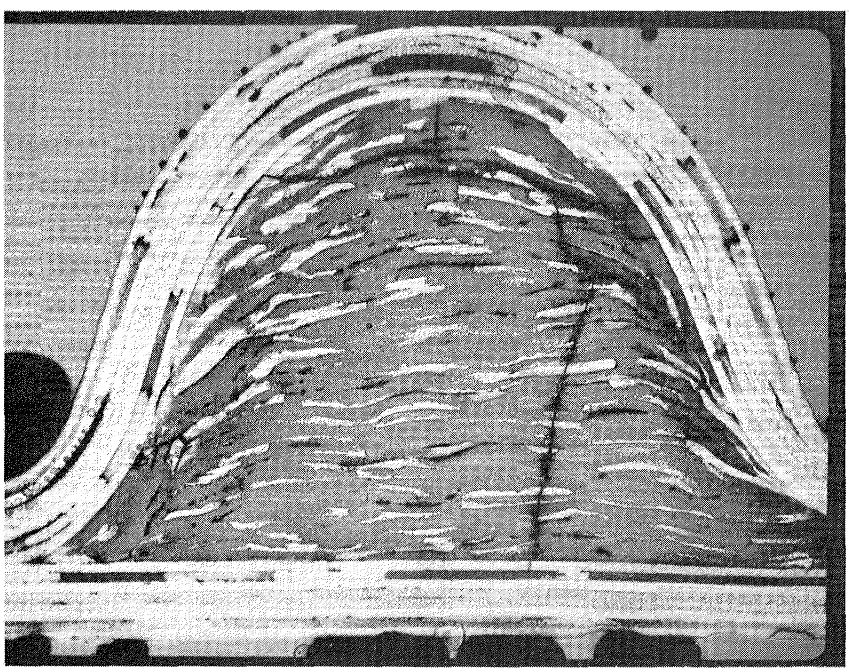


Figure 3-3. Microcracking Through 0 Degree Fibers in "Baseline" Stiffening Bead

Two truss rib caps of the proposed alternate design were molded. One cap was molded with the bead in the female rib cap tool (the tool with the bead recess) and the other with the bead up, that is, the flat side of the web was against the male tool. A flexible nylon vacuum bag was used on the opposite side of the part during cure. Figure 3-4 shows a section of the part which was molded with the bead in the tool cavity and Figure 3-5 shows a section of the part which was molded with the flat side of the web against the flat male tool, with the bead allowed to move under the vacuum bag during cure.

Figure 3-6 and 3-7 show that cracking occurred in both beads, and the beads flattened and spread.

Rib development next progressed to parallel investigations of two alternate rib cap configurations for the truss rib caps.

A "Z" section truss rib was molded using the alternate design concept shown in Figure 3-1 as D. The part was molded "bead-up," that is, with the flat side of the web against the male tool. A molded silicone rubber caul sheet was used against the bead side of the part. Figure 3-8 shows the part side of the caul with the bead cavity molded in. A portion of the completed rib cap is shown in Figure 3-9. The elimination of cracking in the 0 degree fibers, was not accomplished. Figure 3-10 shows that cracking still occurred in the bead 0's and Figure 3-11 shows that cracking also occurred in the 0's which were relocated from the bead to the flange. The distortion in fiber plies visible in Figure 3-11 is due to displacement of the 0 degree ply stacks during laminate layup. Resin content and cured ply thickness was acceptable except in portions of the flange containing 0's, which were outside of tolerance (32.7% resin and 7.0 - 6.8 mils per ply cured thickness).

Another alternate configuration was also considered as a result of the continuing development problems described above and in previous reports. This rib cap design eliminated the bead entirely and simplified the tooling and manufacturing of the cap.


Figure 3-4. Concept "B" Truss Rib Cap with Bead Molded in Cavity

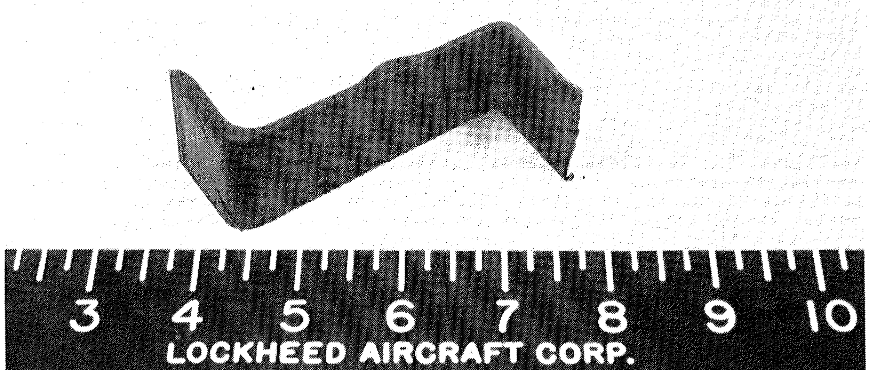



Figure 3-5. Concept "B" Truss Rib Cap with Bead Molded under Vacuum Bag

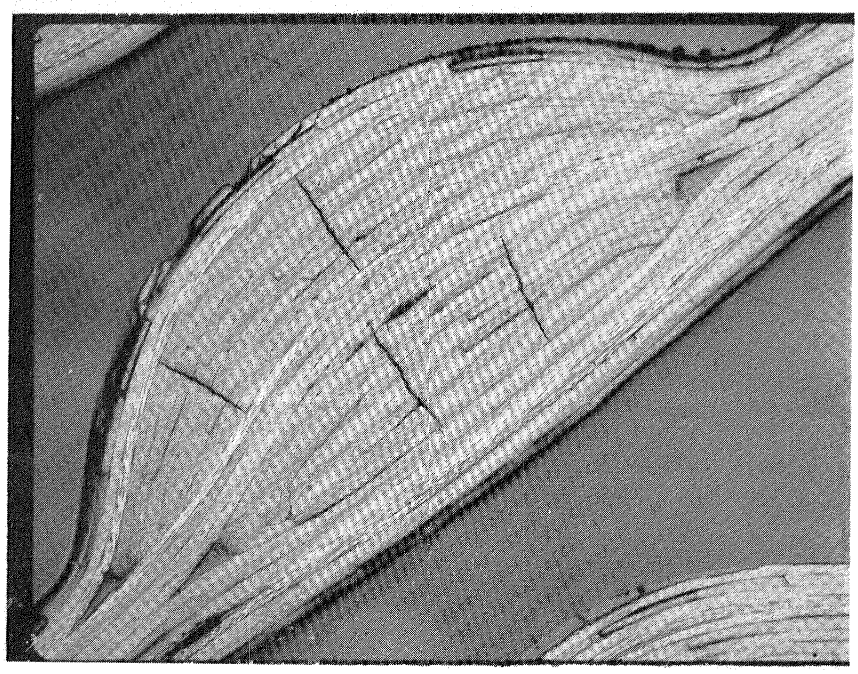


Figure 3-6. Bead Molded in Tool Cavity

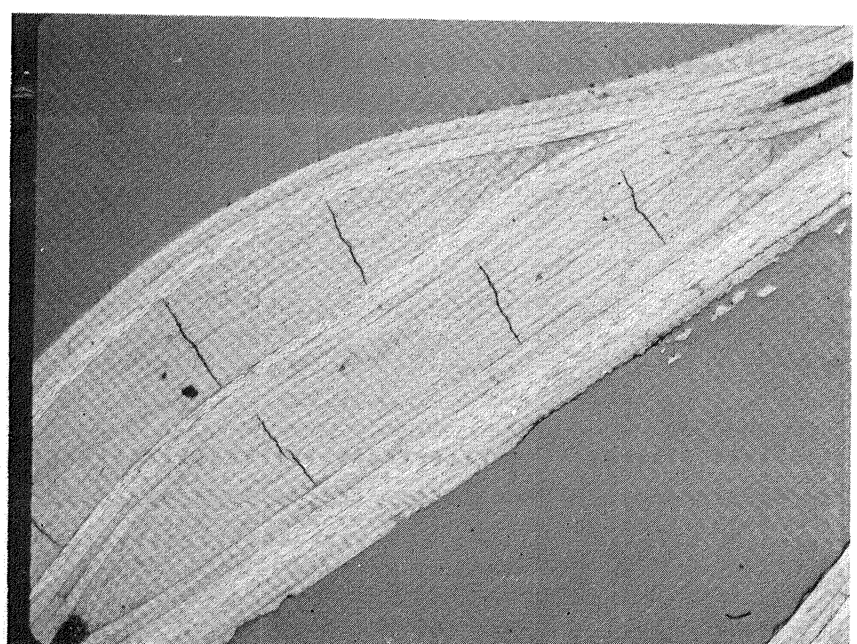


Figure 3-7. Bead Molded Under Vacuum Bag

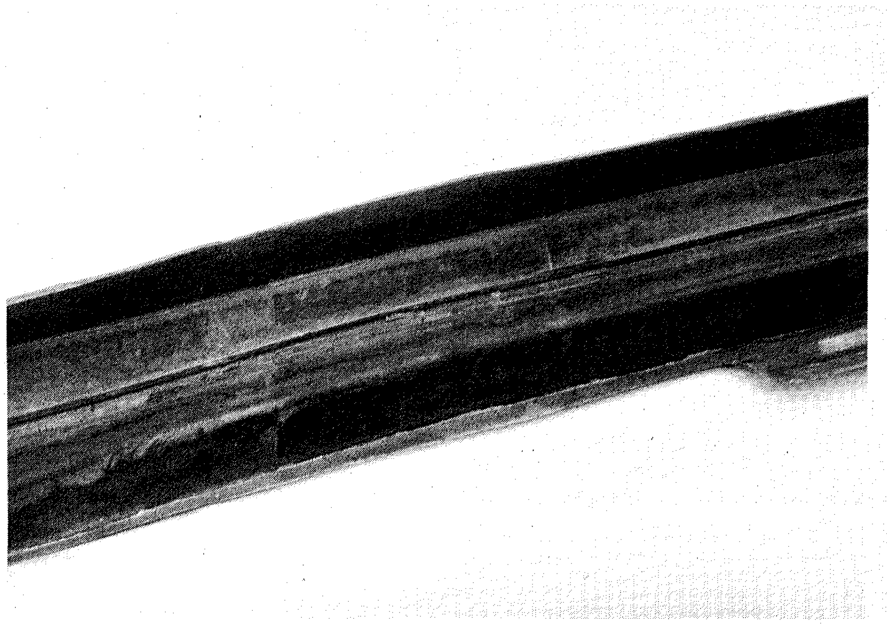


Figure 3-8. Molded Silicone Rubber Caul Sheet used to Mold Concept "D" Truss Rib Cap in Bead-Up Position

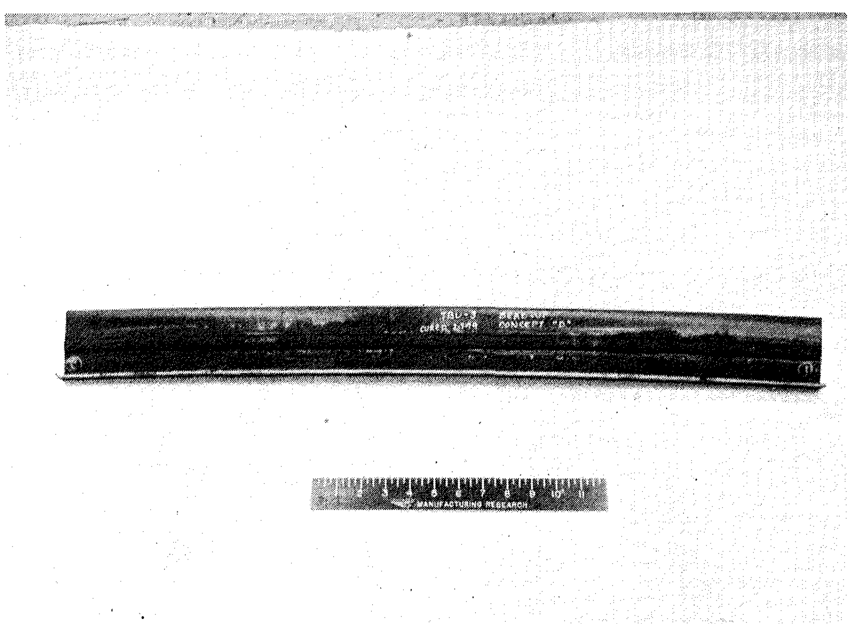



Figure 3-9. Bead Side of Concept "D" Truss Rib Cap

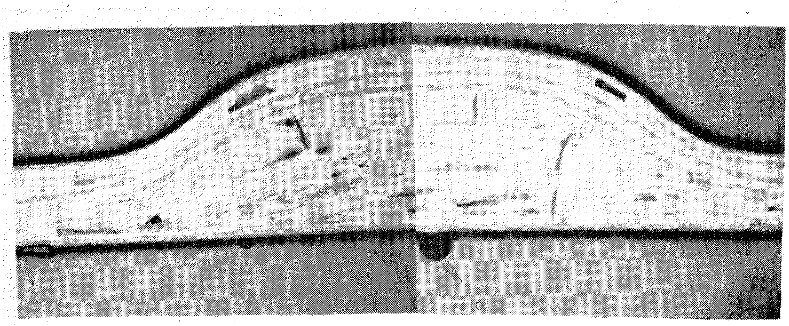



Figure 3-10. Microphotograph of Concept "D" Bead Showing Cracks Through the 0° Plies

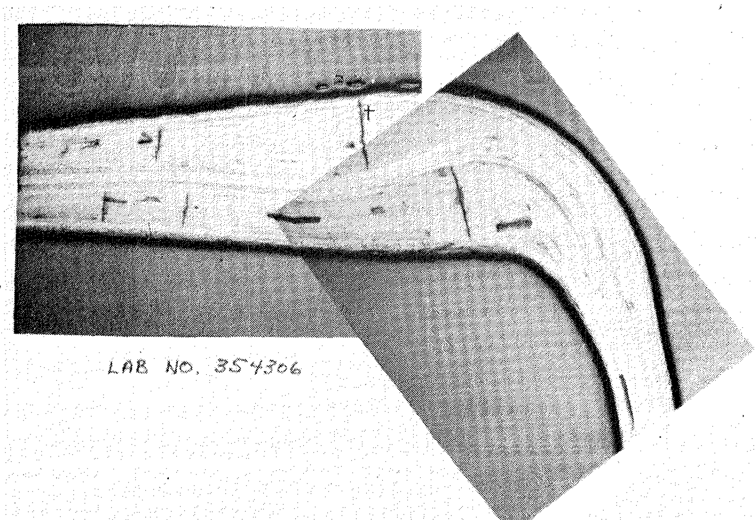



Figure 3-11. Microphotograph of Concept "D" Flange Showing Cracks Through the 0° Plies

Concept E (Figure 3-1) shows the "C" section design to which a development part was first molded during February. A portion of the "C" rib cap is shown in Figure 3-12. The laminate was molded on an aluminum layup block. A formed silicone rubber bag, shown in Figure 3-13 was used as a caul. A nylon vacuum bag over the rubber bag was the pressure diaphragm. Acceptable resin content was achieved.

Process development work continues on the concept "E section" rib cap. A 40 inch long male aluminum block has been made and three ribs caps molded. The bleeder arrangement, 2 plies of peel ply and 1 ply of teflon coated cloth under a rubber bag, was the same for each cap. Three different cure cycles were evaluated; 1) aileron rib cure cycle, 2) ACVF cover cure cycle and 3) a cure cycle suggested by the material supplier (NARMCO). Complete lab results are not yet available, however it appears that satisfactory physical properties can be obtained with all of the cure cycles. Although cured ply thicknesses were within the acceptable range, preliminary lab results indicate low resin contents. These results are being checked by retest. Bleeder arrangement will be modified to obtain satisfactory resin contents.

Figure 3-14. Shows a lay-up and bleeder in place on the development tool. The formed rubber bag will be placed over the bleeder then covered with breather cloth and a vacuum bag. Figure 3-15 shows measurement of the flange of a development rib cap to determine cured ply thickness.

In the baseline solid web rib design, the transverse stiffeners were located in a slotted cavity in the male tool. Fabrication of these "T" sections was difficult and time consuming. They were replaced by bead stiffeners laid on top of the laminate and capped by a 4-ply layer of prepreg. Following a sequence of events similar to the truss rib caps a new bead, designed to reduce microcracking, was developed. This design utilized ±45° plies as well as 0° plies. However, microphotos (Figure 3-16) revealed that microcracks traveled through the 0° plies and were stopped by the ±45 plies.

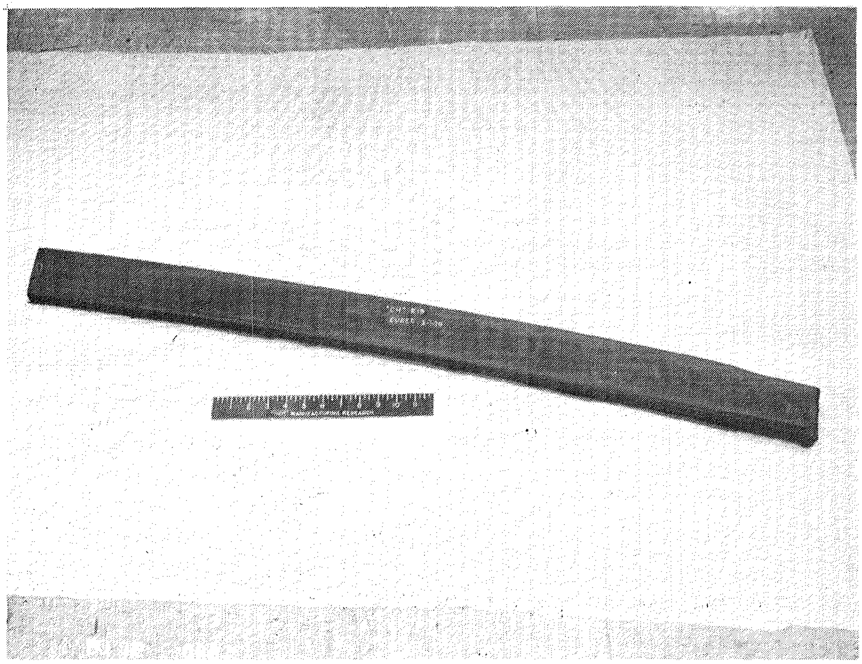


Figure 3-12. "C" Section Truss Rib Cap

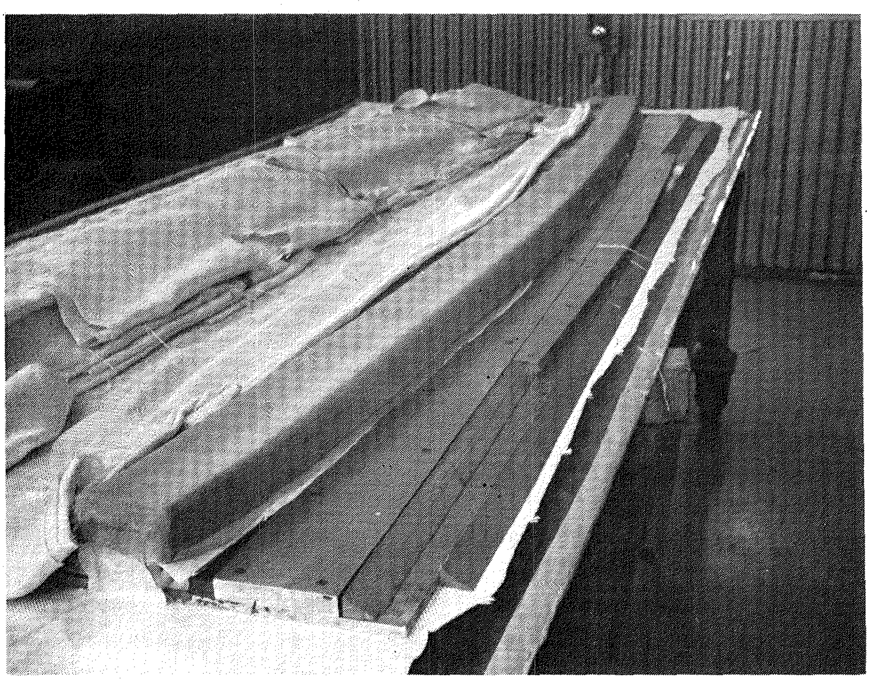


Figure 3-13. "C" Section Truss Rib Cap Layup Covered With Molded Silicone Rubber Caul

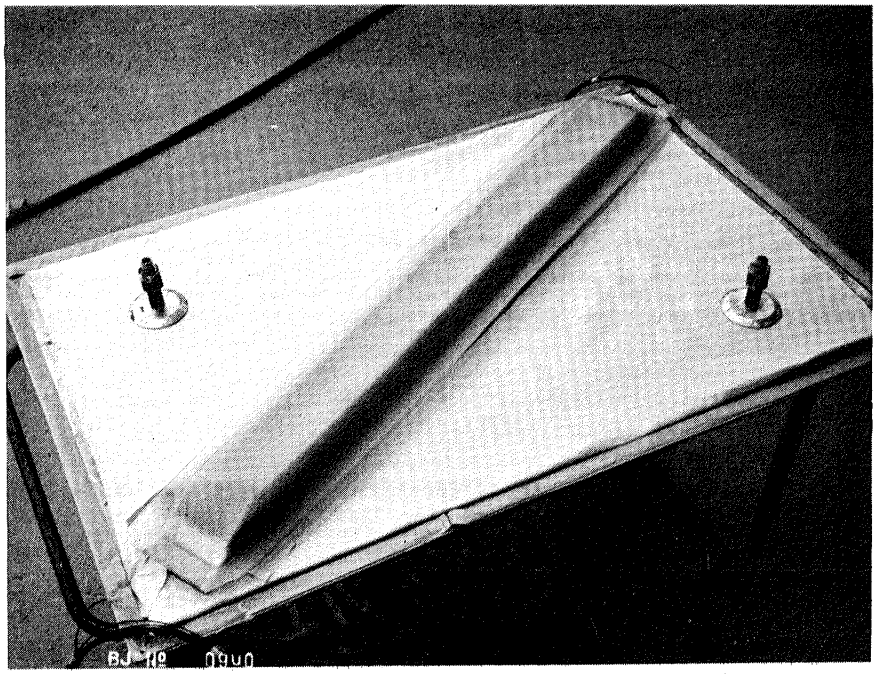


Figure 3-14. Truss Rib Cap Development Tool with Lay-up and Bleeder in Place.

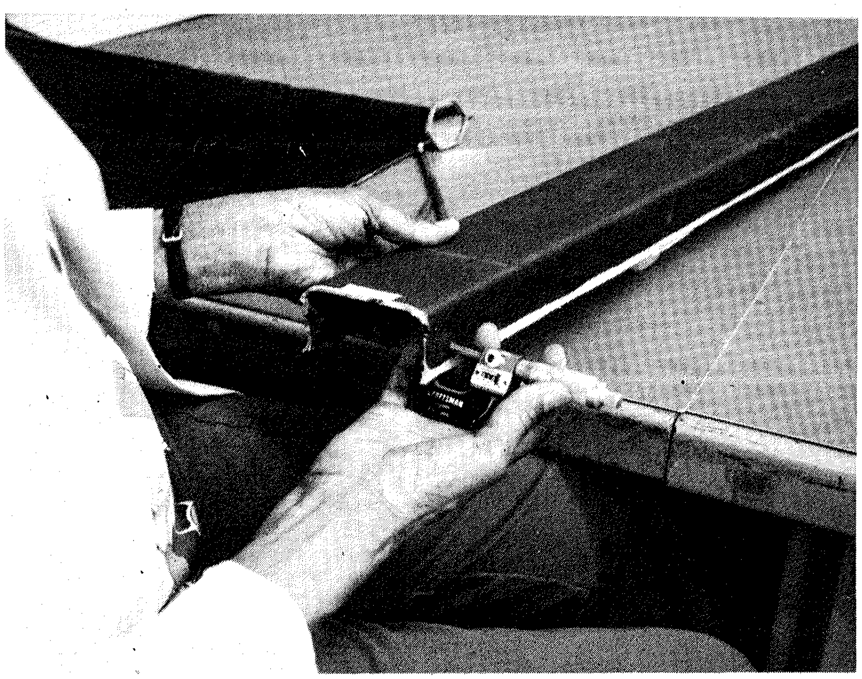


Figure 3-15. Measuring Rib Cap Flange to Determine Cured Ply Thickness

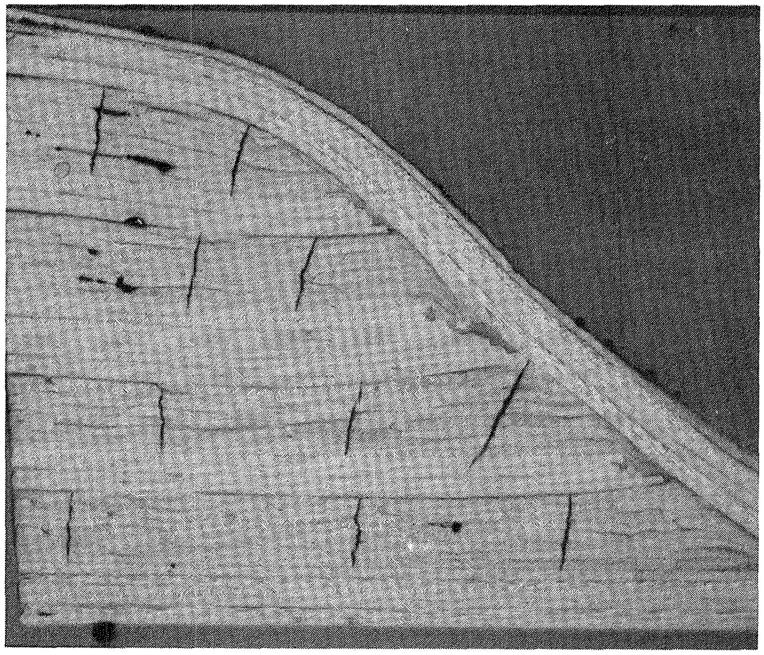


Figure 3-16. Microcracks Through $0^{o}$ Plies of Solid Web Rib Bead Stiffener.

As a result, a new approach to stiffening the solid web rib was taken. Basically the design is a sandwich: six plies graphite tape, syntactic resin core, six plies graphite tape. Figure 3-17 illustrates the basic construction. The syntactic resin core and adjacent graphite preply are cut to size using templates. The part is assembled and "Air Dam" strips are pressed around the exposed edges of the laminate. A molded silicone bag, acting as a caul, is placed over the breather and bleeder system. Finally, a nylon vacuum bag is placed over the entire assembly.

Two development parts were molded to this configuration. The first part is seen in Figure 3-18. Quality Assurance confirmed acceptable resin content, specific gravity, short beam shear, and compressive strength values. An ultrasonic scan detected voids in an area near the flanges next to the trailing edge. Microphotos validate the scan. These voids were caused by the silicone bag which did not fit over the part correctly.

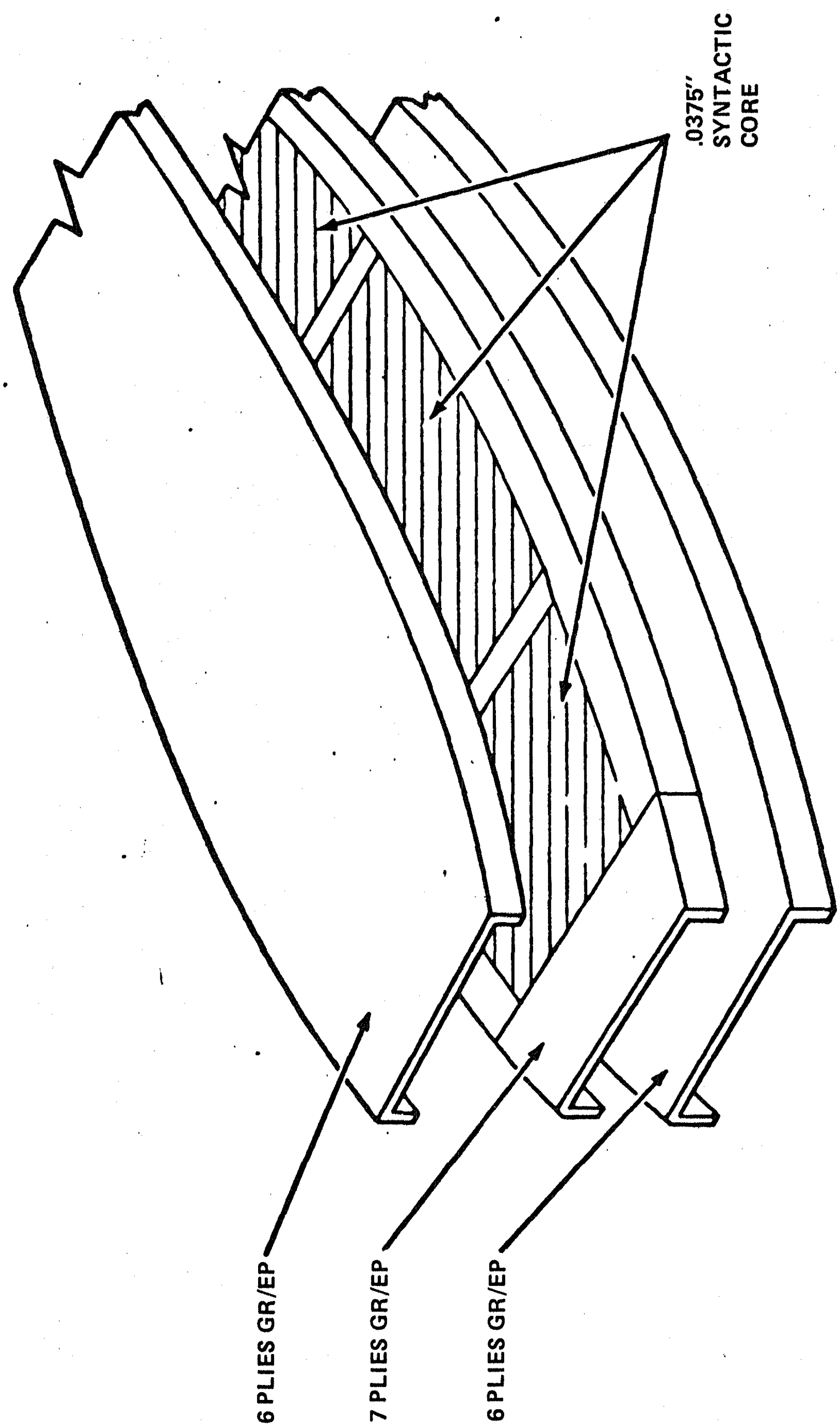



Figure 3-17. Solid Web Rib Basic Construction

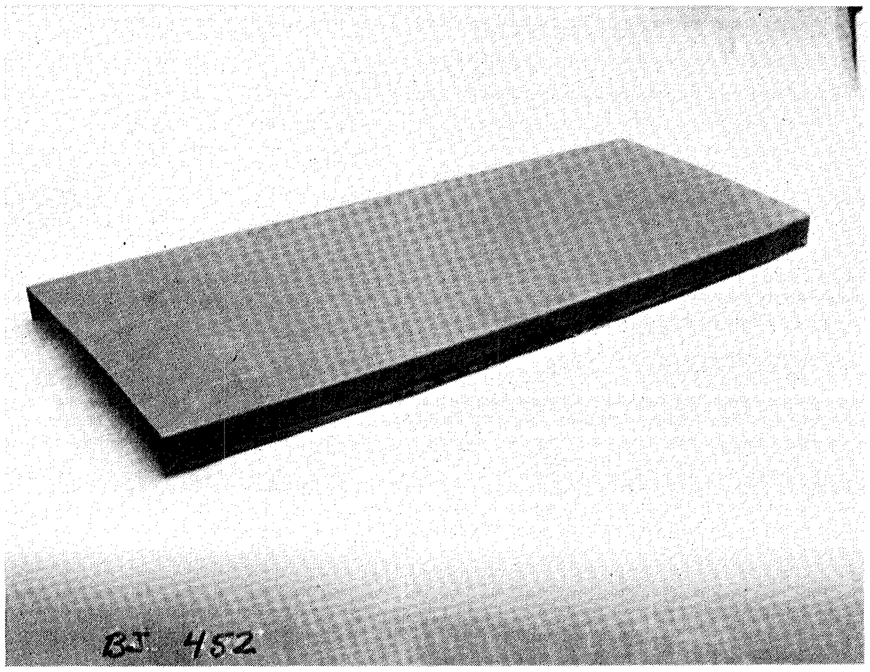



Figure 3-18. Solid Web Rib Made with Syntactic Resin Core

Both parts exhibited some bow and twist after cure. Investigation into the cause and cure of this condition is in progress.

# SECTION 4

# TESTING

Mechanical testing of concept verification specimens was continued. The cover stability test (H27), a static test, and the cover fail safe test (H28), a cyclic load test, were completed during this report period.

## 4.1 COVER PANEL STABILITY TEST (H27)

The objective of this test was to determine initial buckling and ultimate compressive strength of the cover structure after conditioning to 1% moisture absorption. The test was performed at 180°F.

The test specimen size was 24.2 inches wide by 80.0 inches long and consisted of a basic skin panel and three co-cured hat stiffener sections. Intermediate 7075 aluminum angles were installed representing rib station locations. The panel was conditioned in moist air (95-100% RH) at 150°F for 18 days. Traveler coupons run with the panel indicated that the panel had a moisture content of 1.01% after this exposure. The panel was instrumented with ten axial-type strain gages installed as schematically shown in Figure 4-1.

The test panel was then installed in the Baldwin 400K static test machine as shown in Figures 4-2 and 4-3. An environmental chamber was constructed around the test panel complete with its own moisture generator and heater/circulation fan unit. Preliminary runs were conducted to check the specimen alignment in the test machine and operation of the environmental system. The environmental chamber, with the specimen installed, is shown in Figure 4-4. The moisture generator provided 95-100% relative humidity and the heater/circulation unit brought the panel surface up to the desired 180°F at the 95-100% relative humidity.

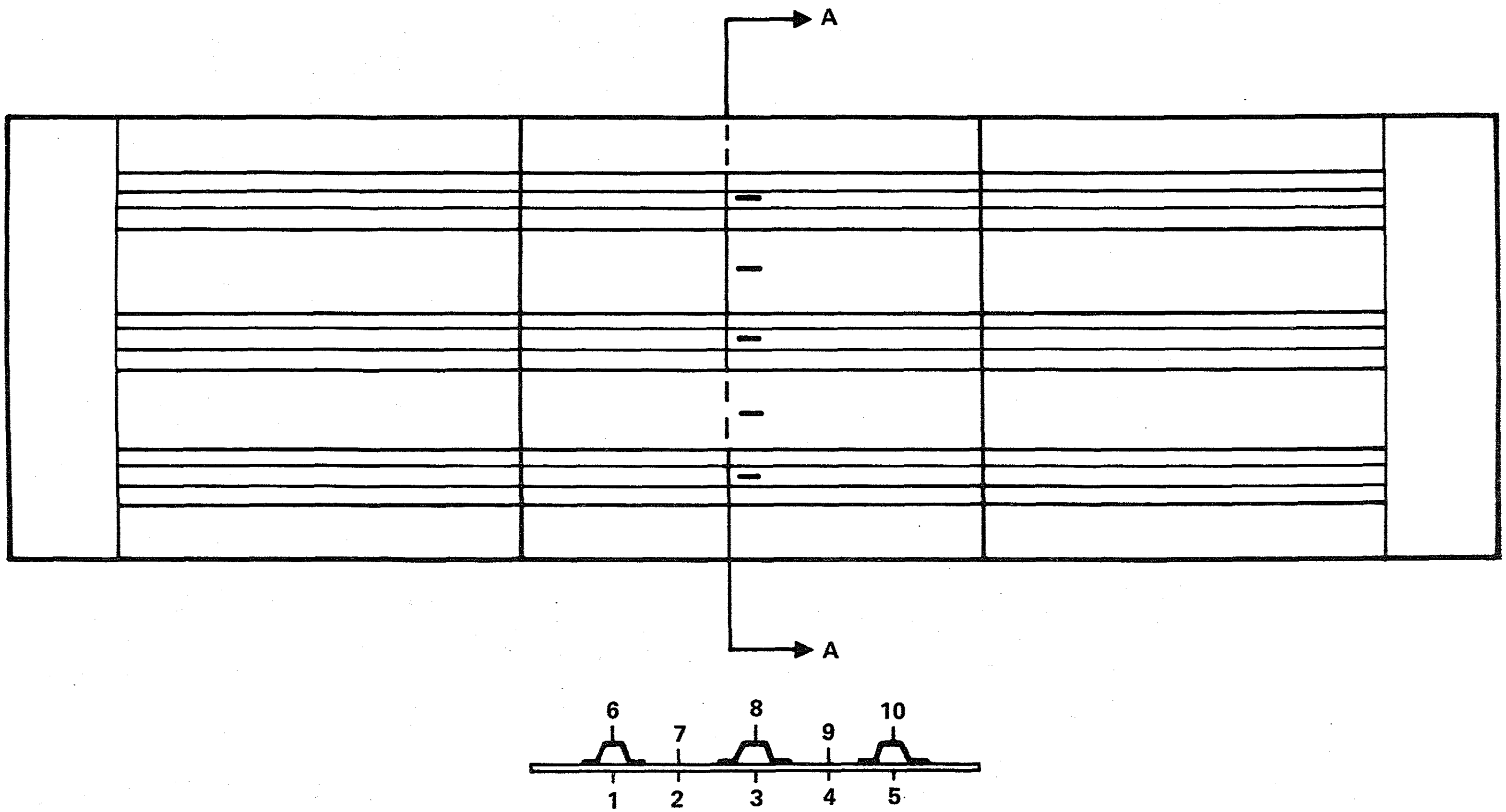



Figure 4-1. Schematic of Strain Gage Location for H-27

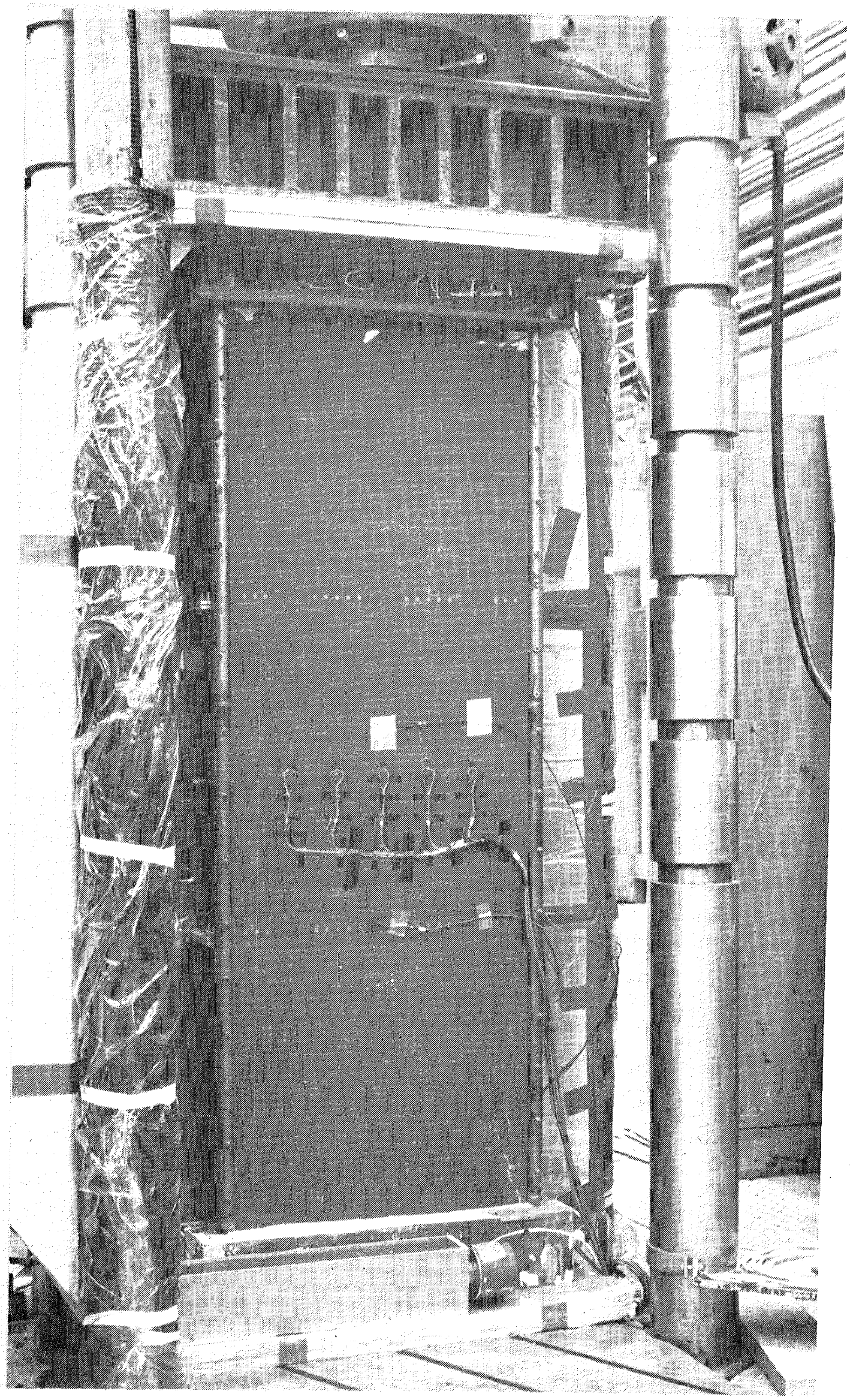


Figure 4-2. Test Setup

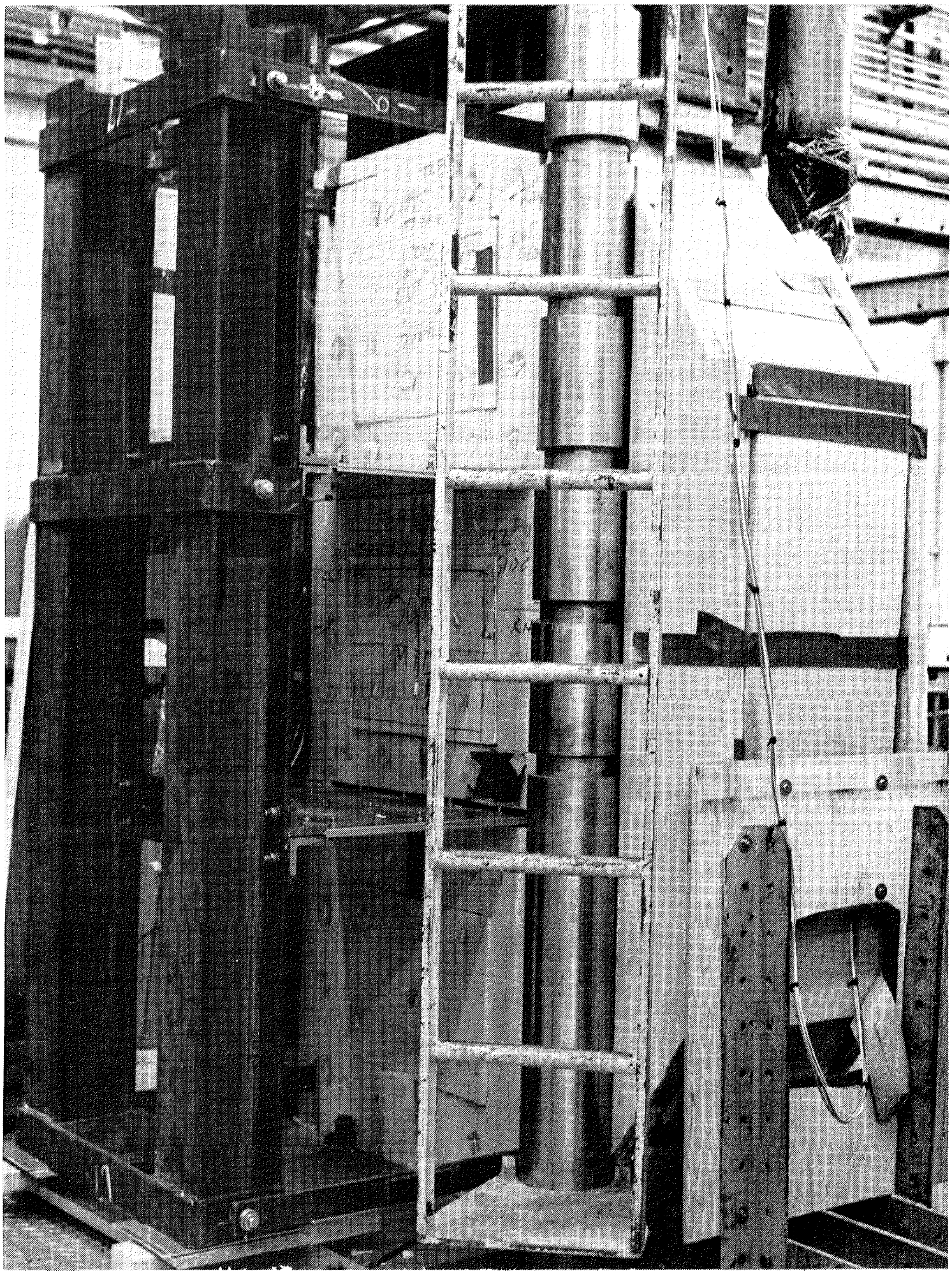


Figure 4-3. Setup Showing Rib Flexures Protruding from Chamber

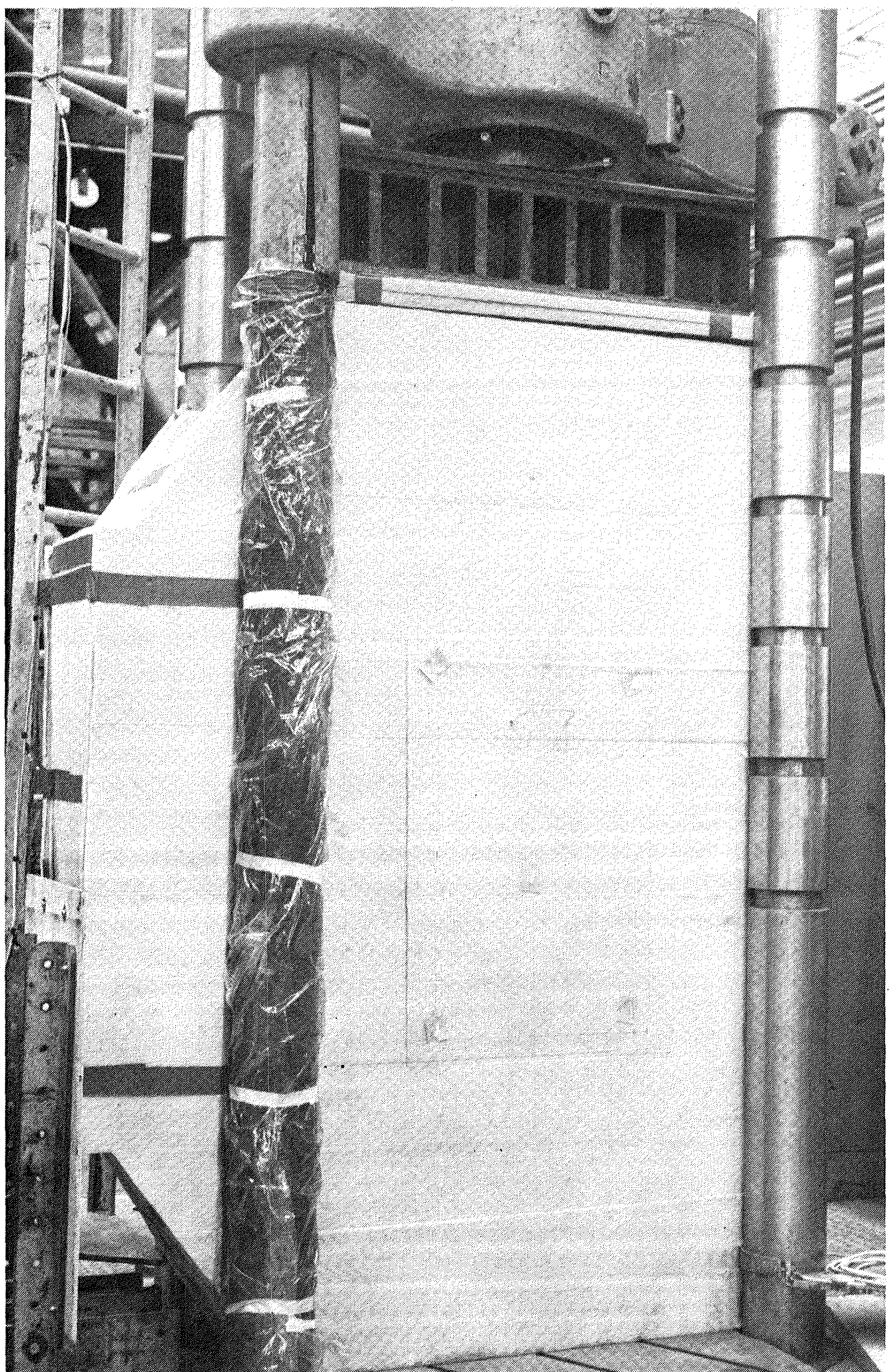


Figure 4-4. Setup Showing the Environmental Chamber

The test was conducted in two parts. First, the panel was loaded up to design ultimate compression load of 58,100 lbs with continuous data recordings. The panel load was reduced to 10,000 lbs and held while the recorded data was examined. The panel did show some initial buckling in the skin at 54,000 lbs load, as can be seen from Figure 4-5.

The second test objective was to run the panel continuously to failure with the data being continuously recorded during the test. Failure of the panel occurred at 70,000 lbs, 120% of ultimate load. Initial buckling was indicated in the back-to-back skin gages at about 54,000 lbs compression. Figures 4-6 and 4-7 show the buckling of the skin panels between the hats. The predicted failure loading was 80,000 lbs and predicted initial buckling was 48,000 lbs.

Failure occurred in the lower bay of the 3-bay panel. Examination of the panel indicated that a skin/hat separation had occurred with subsequent rapid failure of the hats and skins shown in Figures 4-8 and 4-9. The failure is considered to be fixture induced as the end potting introduces a bending moment due to the built in effect. The damaged area will be cut off and the specimen will be retested in a different configuration to find the true failure load.

## 4.2 COVER FAIL SAFE TEST (H28)

Concept verification Test Item H28 had the objective of establishing the crack propagation and fail safe behavior of the hat stiffened cover design when exposed to flight-by-flight spectrum loading. The obviously damaged panel was required to withstand a half-lifetime of spectrum fatigue, including limit load. ~~with obvious damage~~.

The panel was a five hat stiffened section approximately 41 in. wide and 80 in. long. The test section had two simulated rib stations attached and had end attachments for test machine attachment as shown in Figure 4-10. A cut was placed on the face sheet, 2.25 inches long, directly under the middle hat as shown in Figure 4-11a. A photograph of the saw cut panel is shown in Figure 4-12. The panel was instrumented with two axial strain gages on the

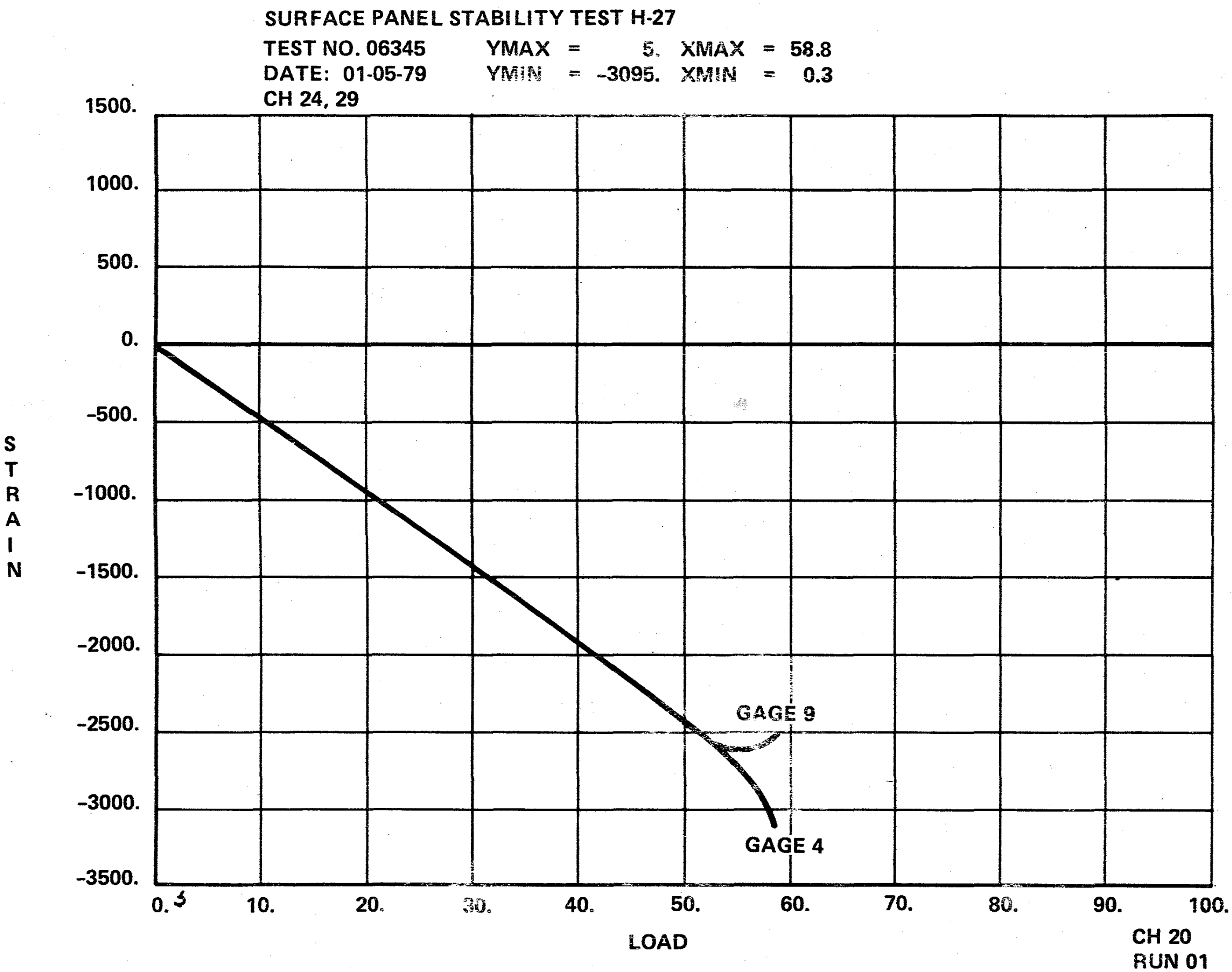



Figure 4-5. Strain Gages 4 and 9 - Limit Load Condition

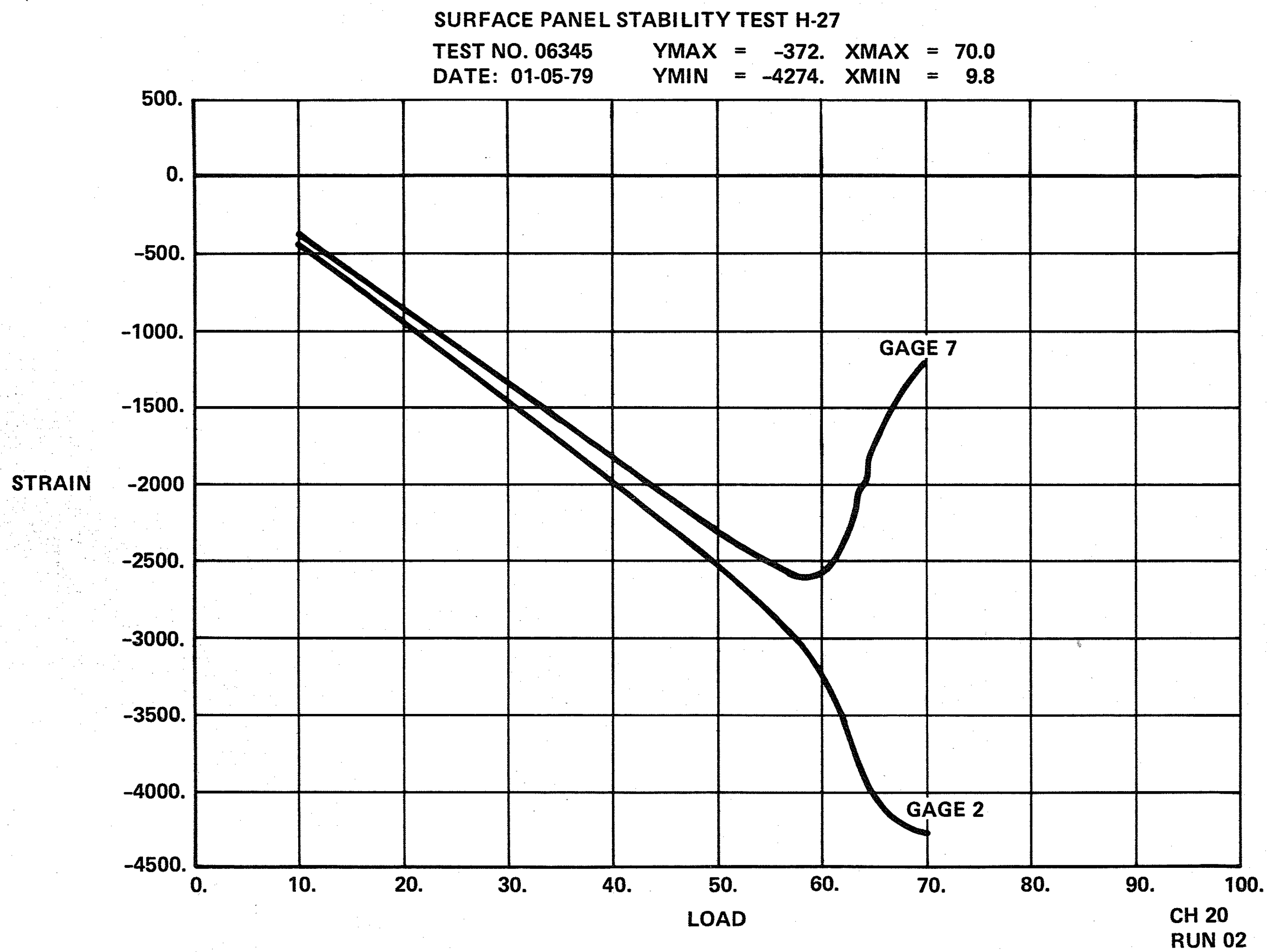



Figure 4-6. Strain Gages 2 and 7 - Failure Condition

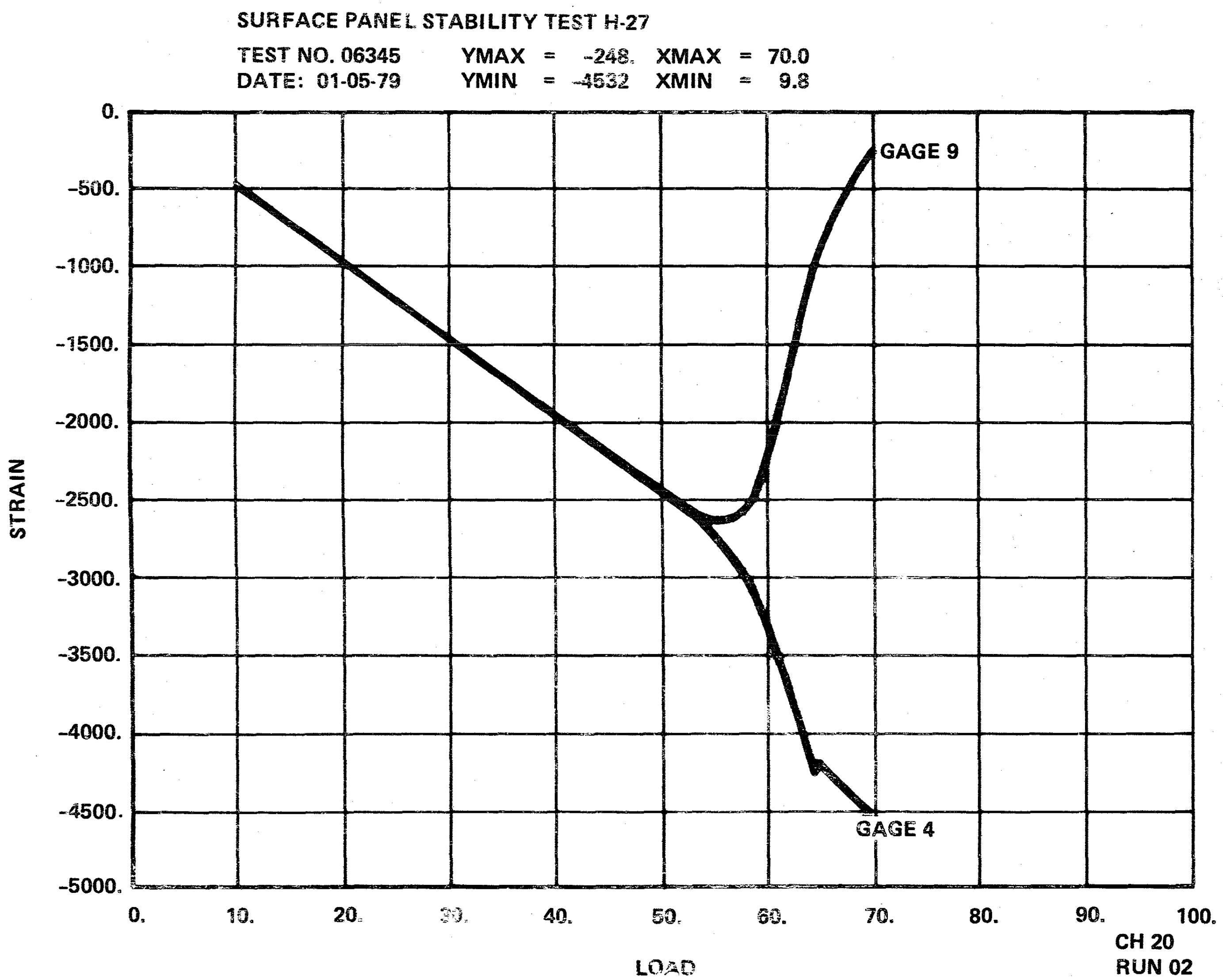



Figure 4-7. Strain Gages 4 and 9 - Failure Condition

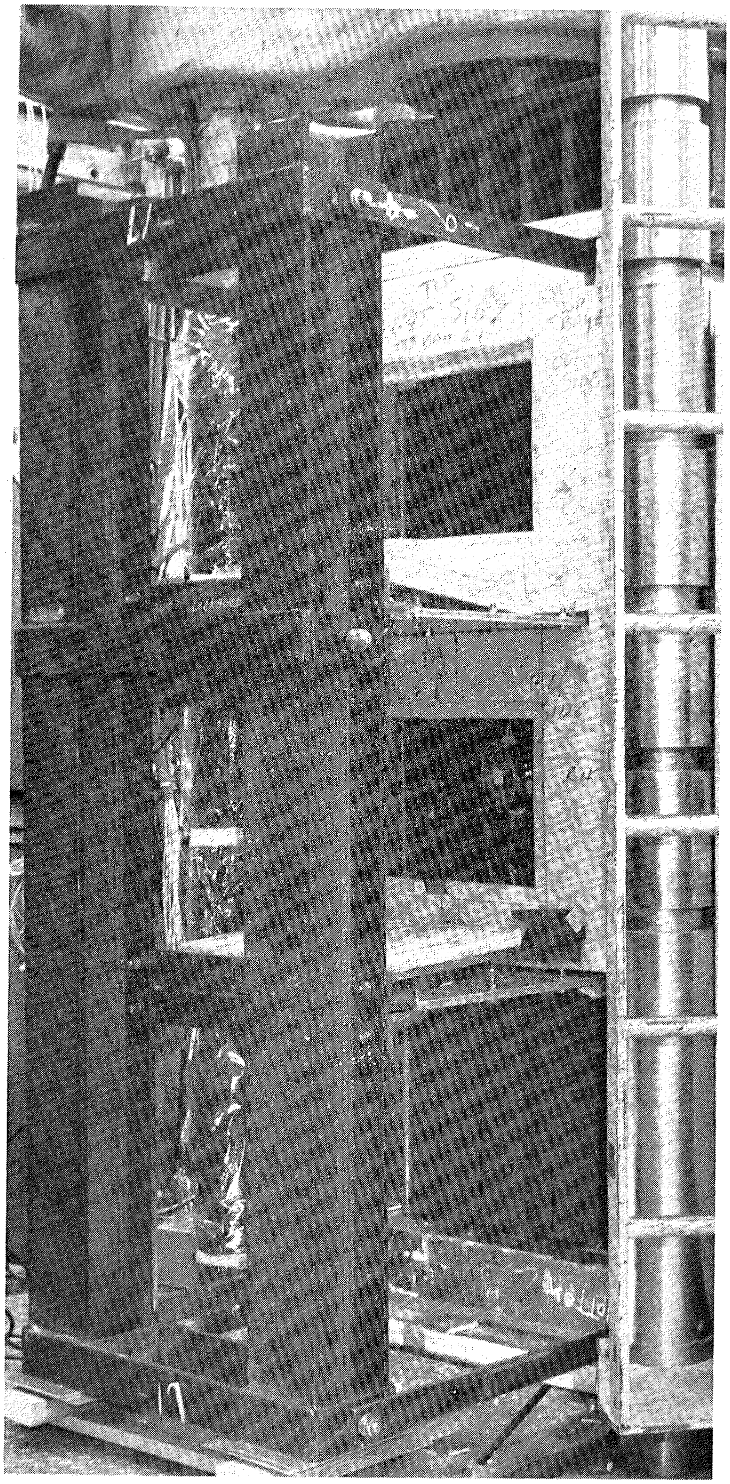


Figure 4-8. Shows Failure in the Lower Bay Area

Figure 4-9. Closeup of Skin Side Failure

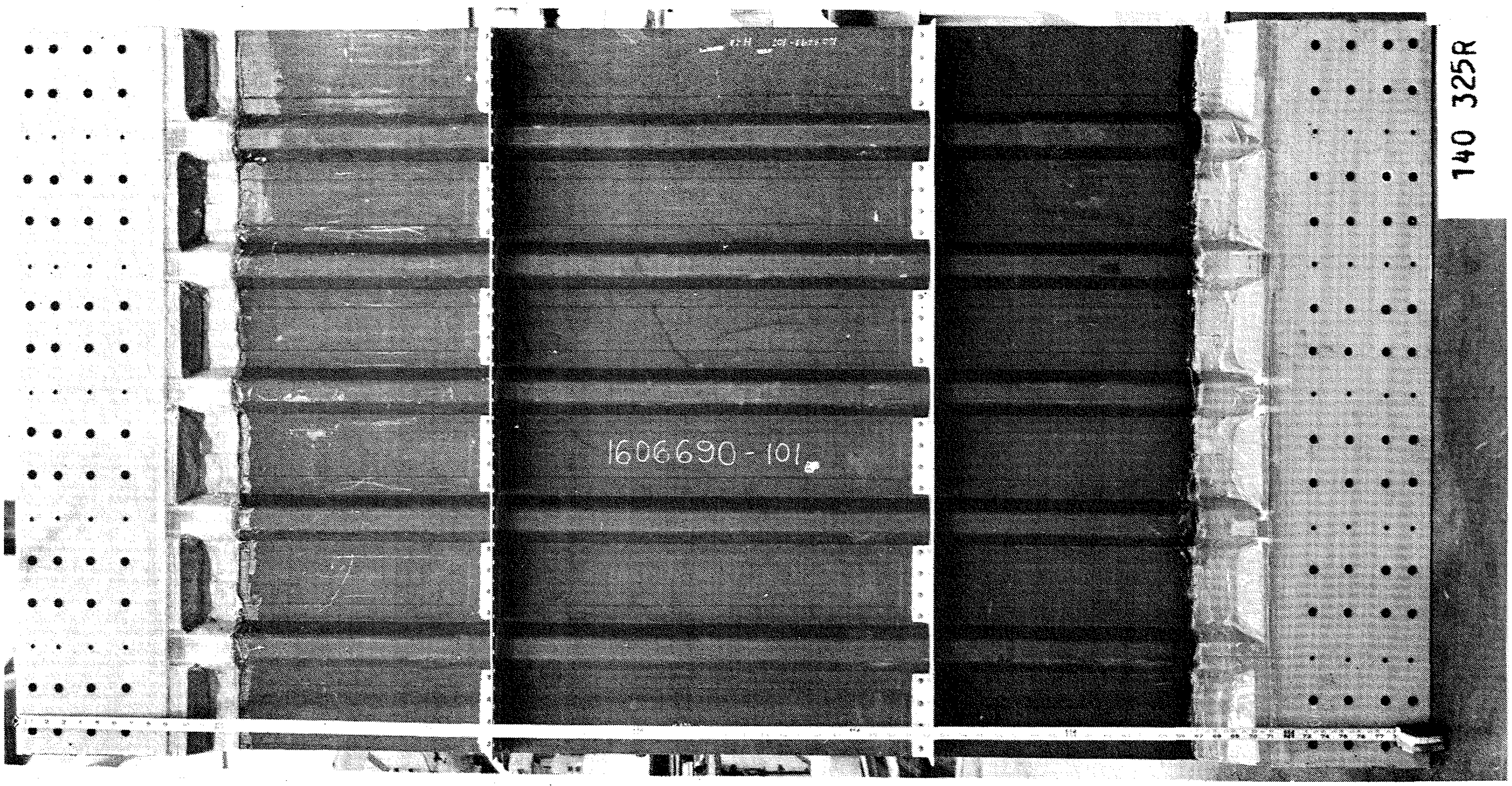



Figure 4-10. Test Panel H28 Assembled for Installation in Test Machine

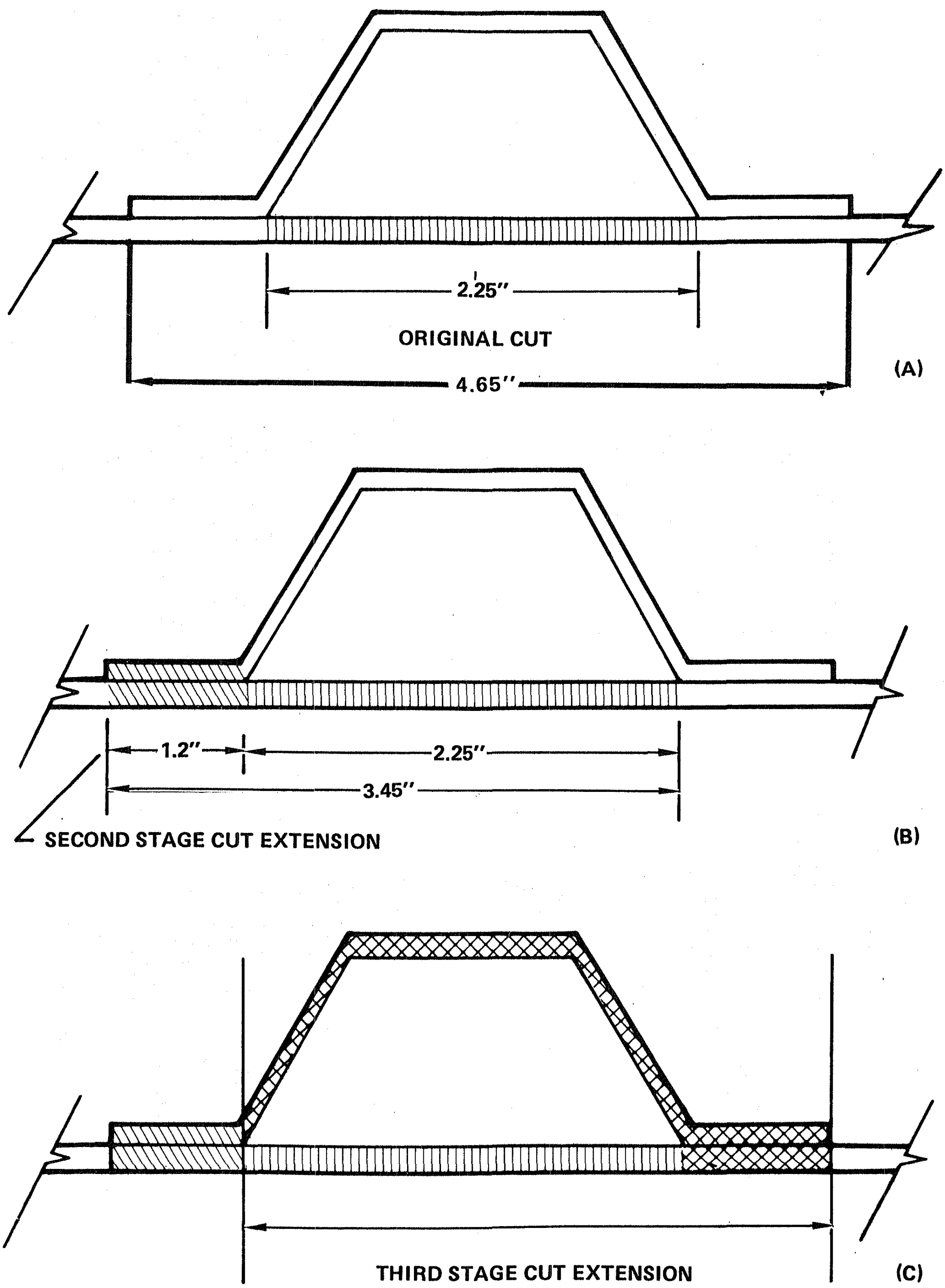



Figure 4-11. Panel Saw Cut Stages

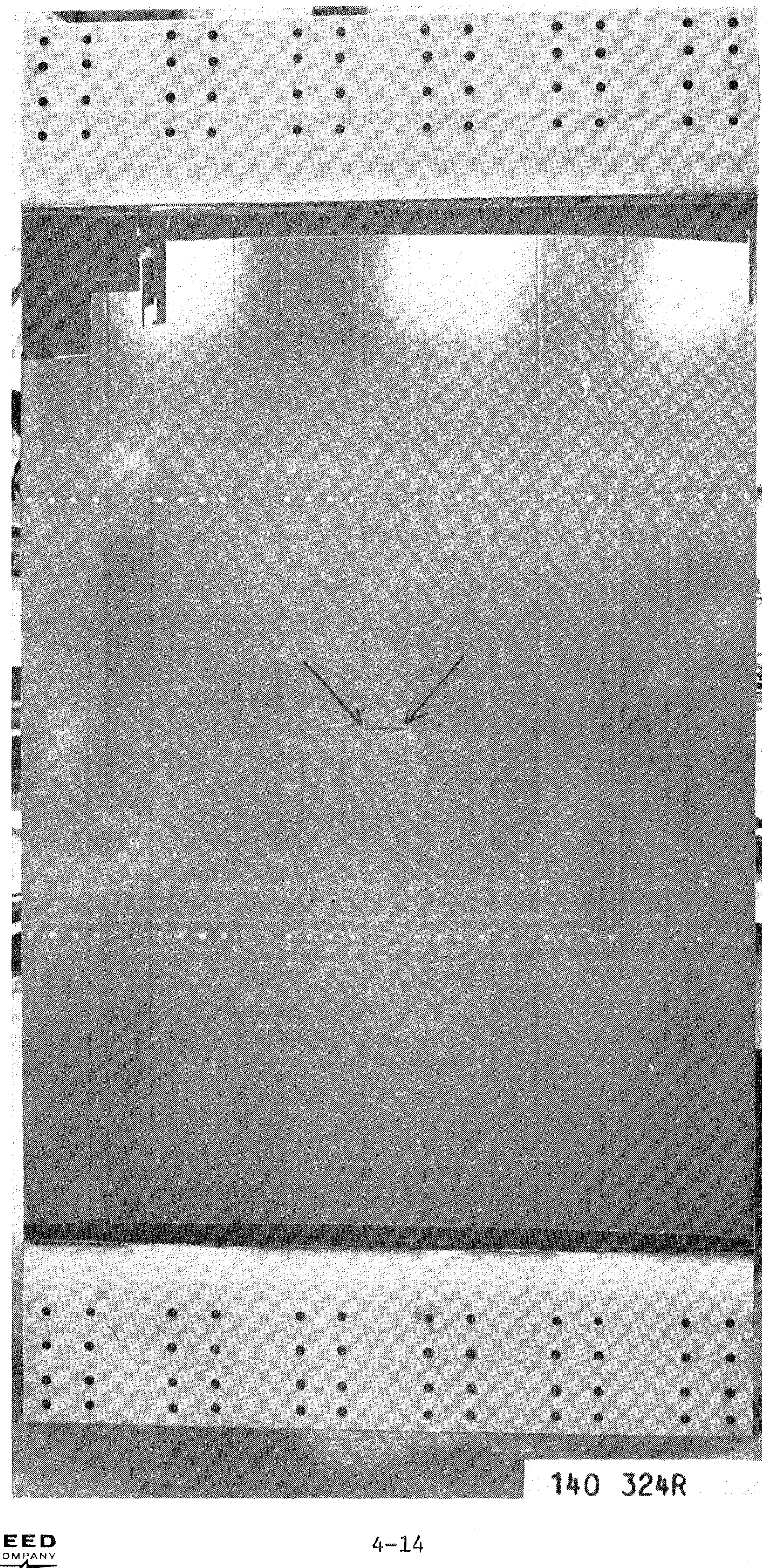



Figure 4-12. Saw Cut Under Center Hat

hat caps in line with the initial saw cut as shown in Figure 4-13 and six gages on the skin face as shown in Figure 4-14.

The panel was installed in the MK II horizontal test machine as shown in Figure 4-15 for flight-by-flight spectrum testing . After completion of 9000 flights (1/4 lifetime), the saw cut region was ultrasonic inspected and no crack growth defected. The panel was then subjected to the limit load cycles and again ultrasonically inspected. No damage was detected. The saw cut was then extended to 3.45 in. as shown in Figure 4-11b which included one side of the flange region. Flight-by-flight spectrum testing continued for an additional 1/4 lifetime. During this testing sequence, some small cracks were noted in the resin region at the hat flange/skin interface, but no growth from these cracks occurred. Further ultrasonic inspection did not indicate any subsurface deterioration.

After the 1/2 lifetime (18,000 flights) the limit load cycles were applied. A loud "pop" was audible and visual examination showed that cracks had developed in two areas: (1) at the end of the saw cut in the skin, 3 cracks 1/8 to 3/16 inches long;and (2) in the hat side wall at the region of the saw cut intersection. The skin cracks were examined ultrasonically and were found to be surface delaminations only. The hat cracks and damage were confined to the side wall of the hat in an area about 3/8 inches in diameter. The flight-by-flight spectrum was again initiated for another 1/4 lifetime. During the first 1800 flights of this sequence, the crack on the hat sidewall grew to the crown of the hat and stopped. No further growth was noted for the duration of this run other than some minor surface layer fiber delaminations. Also noted were some minor amounts of surface fiber lifting at countersunk fasteners between Bays 2 and 3. The spectrum testing was continued for an additional 1/4 lifetime (full lifetime total) with no change in growth noted. Up to this time 2 cycles of limit load had been applied to the specimen.

After completion of the full lifetime (36,000 flights), the saw cut was extended completely through the hat including the crown and both flanges as shown in Figure 4-11c. The spectrum loading was then continued for an additional 1/2 lifetime. Only minor crack growths were noted.

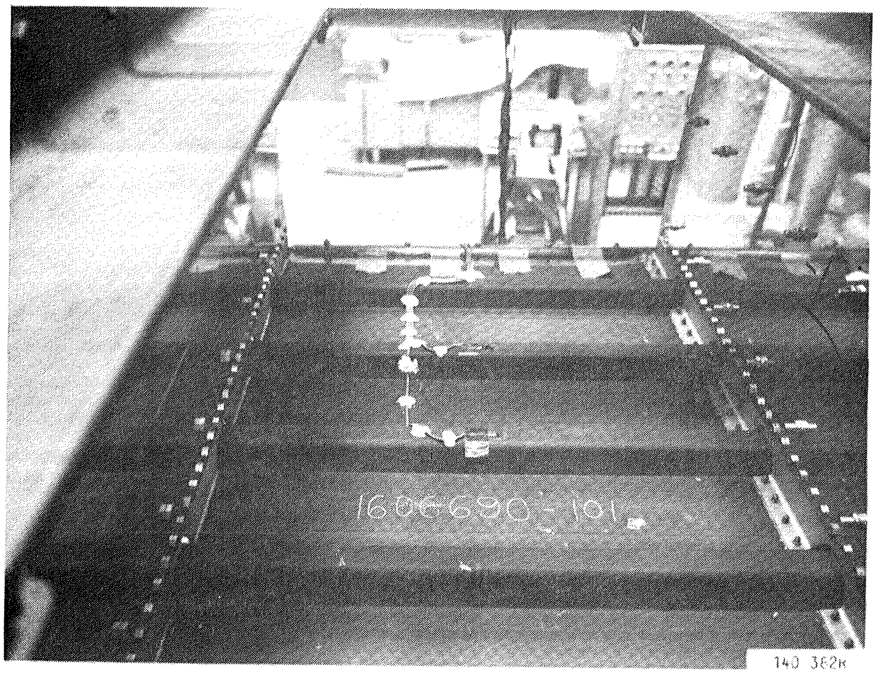



Figure 4-13. Strain Gages on Stiffener Side

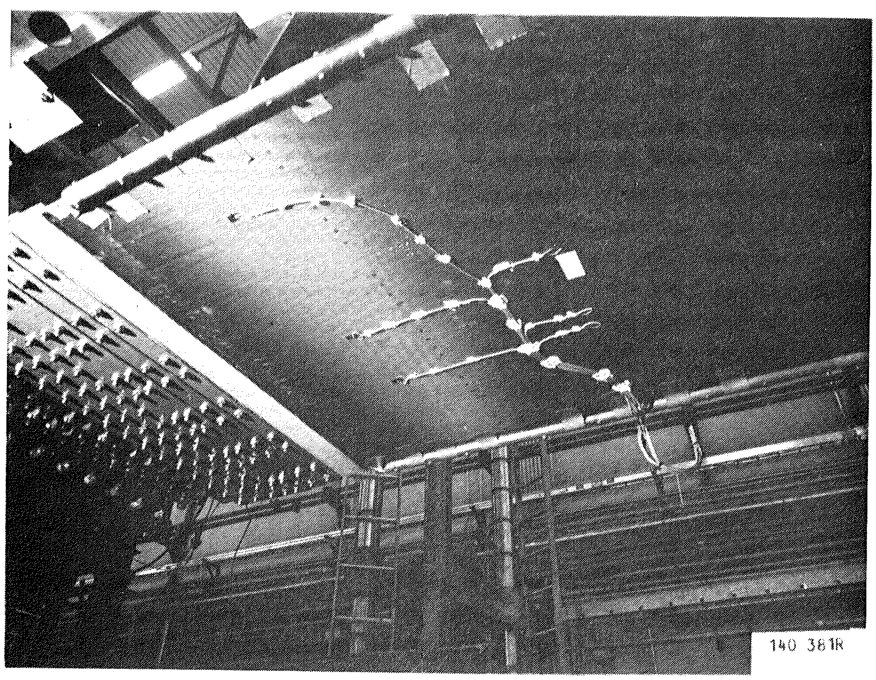



Figure 4-14. Strain Gages on the Skin Side

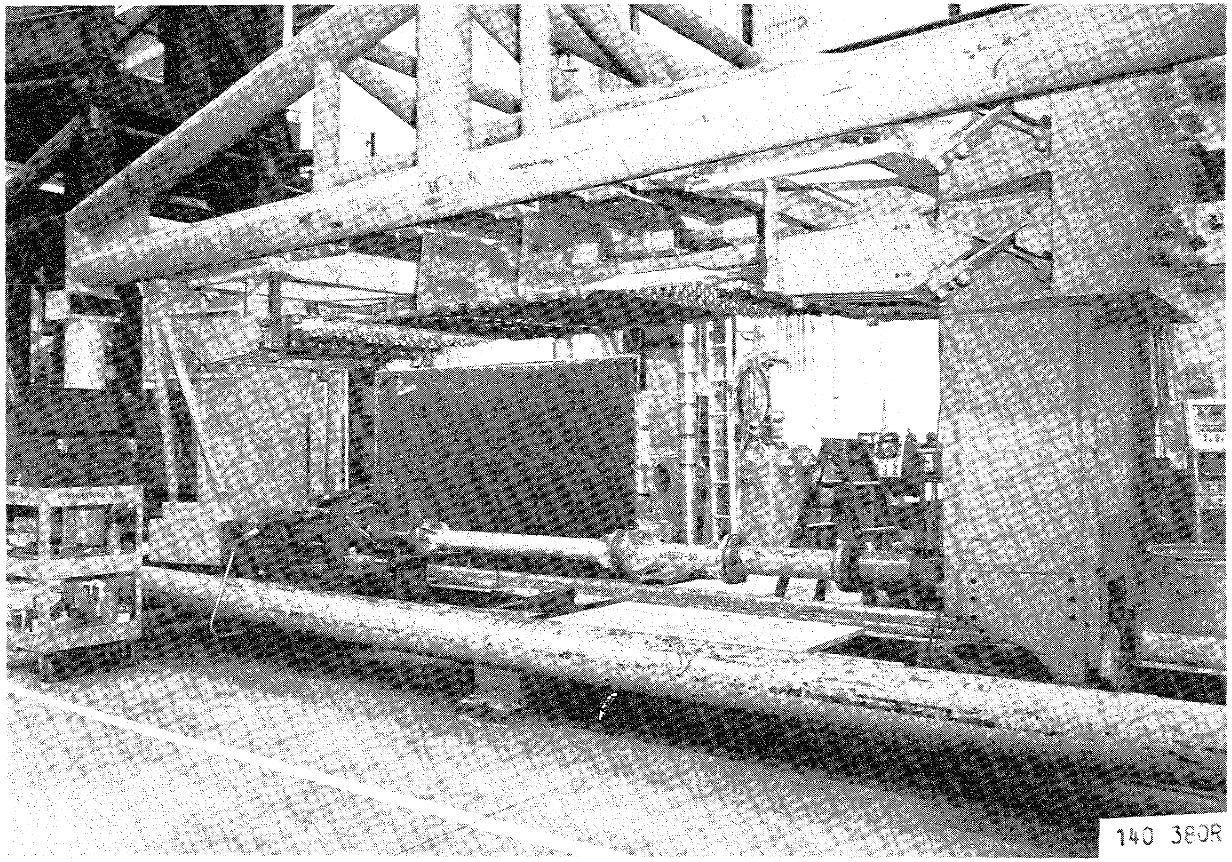



Figure 4-15. H-28 Panel Installed in the Test Machine

After completion of this sequence, the limit load cycles were again applied. Failure of the panel occurred at 96% of limit load in compression. Figures 4-16 and 4-17 show the panel after failure. The panel had carried 100% of limit load in tension and was in the limit load compression cycle at failure.

The test demonstrated that the cover can withstand one-half lifetime of fatigue loading and can carry limit load with obvious partial damage. The panel in fact sustained one-and-a-half lifetimes of fatigue and two applica-cations of limit compression and three applications of limit tension. Failure in the compression mode was due to the high stress concentration at the end of the saw cut which was a tenth-inch wide. Local surface delaminations at these ends precipitated failure.

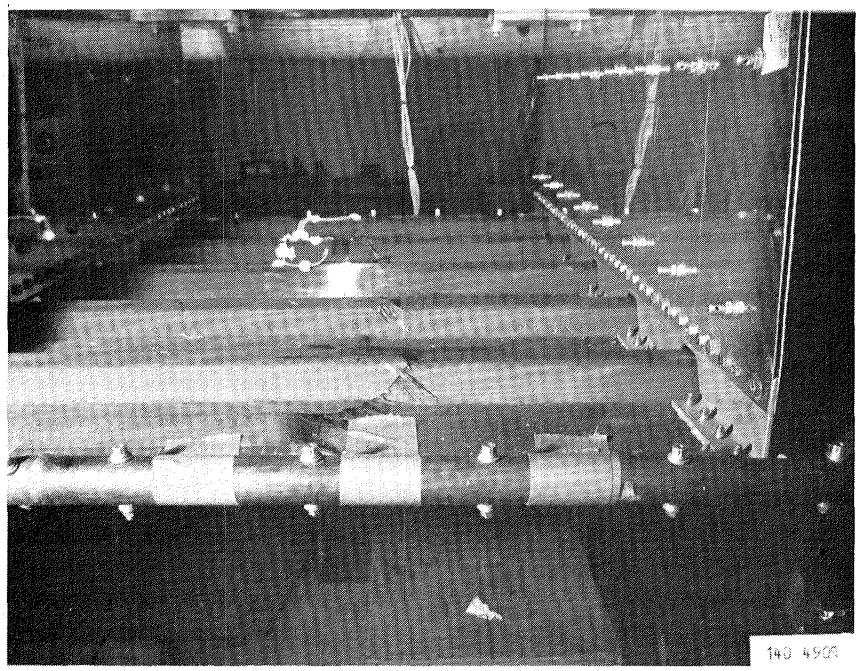


Figure 4-16. Stiffener Side After Failure

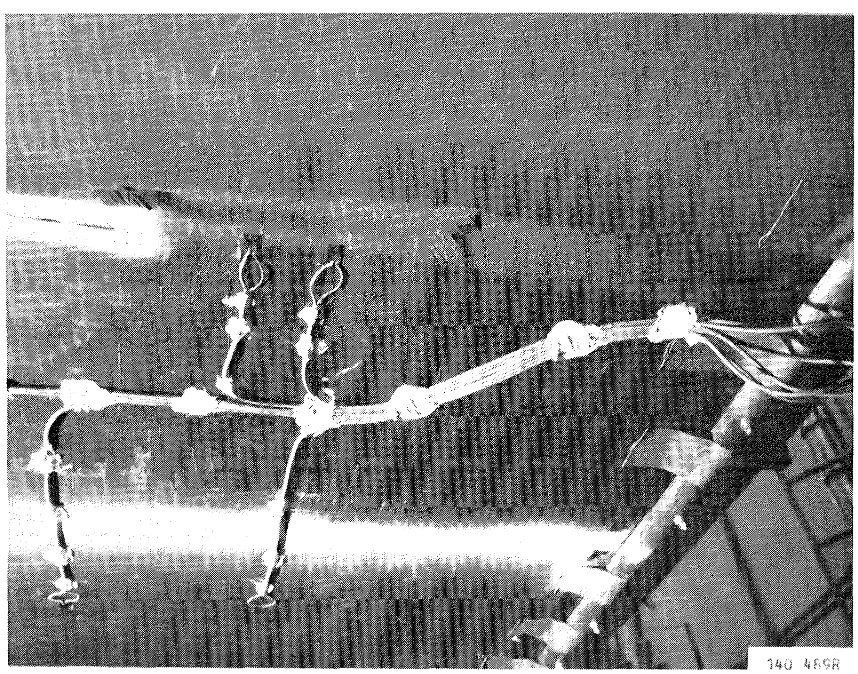


Figure 4-17. Skin Side After Failure

# SECTION 5

# PRODUCTION READINESS VERIFICATION TESTS

The ACVF program does not include flight service evaluation but alternately provides for testing of multiple large-scale subcomponents of the structure for evaluation of variability in static strength and for assessment of durability under extended-time laboratory tests involving both load and environment simulation. The production readiness verification program (PRVT) is supplemental to the ancillary test program. These tests are designed to provide information to answer the following questions:

- What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality levels and quality control procedures?
- What variability in static strength can be expected for production quality components, and are the margins sufficient to account for this variability?
- Will production quality components survive extended time laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence of in-service durability?

The questions are not primarily directed towards basic material properties. It is believed that the combination of service experience on secondary structures and coupon tests in the ancillary test program provide confidence in durability of the basic material. The questions are directed instead to the realities of production quality as influenced by cost objectives and by scale-up and complexity effects which will cause structural quality to differ from that represented by idealized small coupons.

On each of two key structural elements of the ACVF, ten static strength tests and ten durability tests will be conducted. One element will represent the front spar/fuselage attachment area, and the other element will represent the cover/fuselage joint area.

## 5.1 FABRICATION

PRVT cover panel units 13 through 20 were laid up and cured during this report period. Figure 5-1 shows the layup operation in progress on a typical pair of PRVT cover assemblies. Figure 5-2 illustrates typical root end configuration and tooling details for the hat stiffeners before application of bleeding stacking. In Figure 5-3 all hat stiffeners have been located on each skin. In the next operations the cauls will be removed to permit location of one ply of peel material and one ply of barrier film over the hat stiffeners. Figure 5-4 shows the layup complete with all bleeder plies in position over the entire layup. The parts are ready for replacement of cauls, installation of breather plies, vacuum bagging, and cure.

A change in bleeder stacking was used on PRVT units 17 through 20 to obtain some units which would have resin content between 30 and 32% so that the high end of the allowable range can be tested. The first 16 were all below 30%. It was concluded this could be accomplished by elimination of the single ply of 120 cloth bleeder running the full length in the skin area between the hat stiffeners. In addition, all plies of nexus bleeder in the 34 ply skin buildup were also removed. Aside from these changes, the bleeder stacking remained unchanged from the bleeder stacking used on previous PRVT panels.

## 5.2 SPAR DURABILITY

During this quarter all ten spars have been assembled into test box structure pairs, instrumented, and installed in the test chambers as shown in Figures 5-5 and 5-6. One spar in each chamber is instrumented with four axial and six rosette strain gages, as shown in Figure 5-7; the remainder have only one rosette (#5) on each spar. All strain gages have been connected and are being calibrated. All load trains have been installed but not connected to the specimens. Checkout of the computer control system is

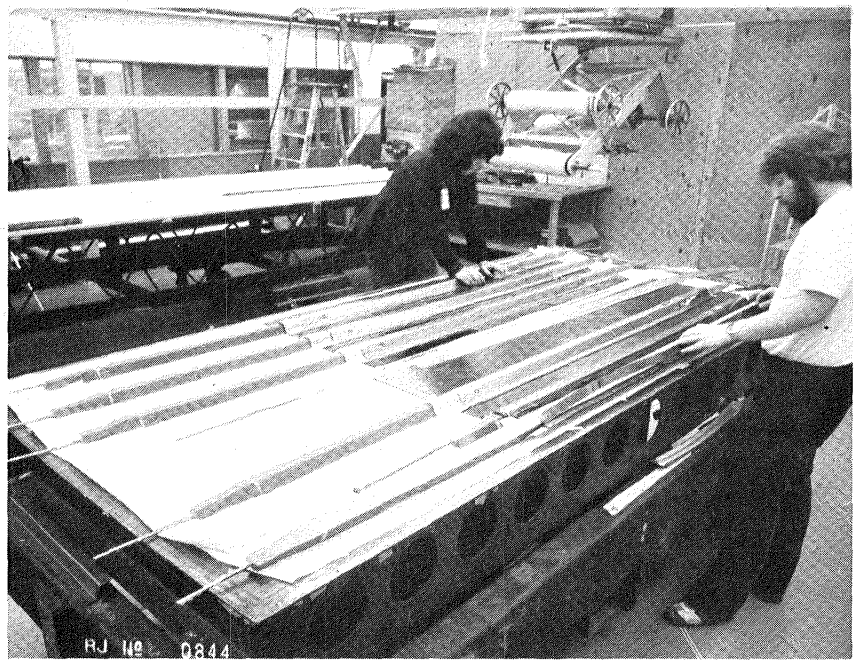



Figure 5-1. PRVT Cover Panel Layup in Progress

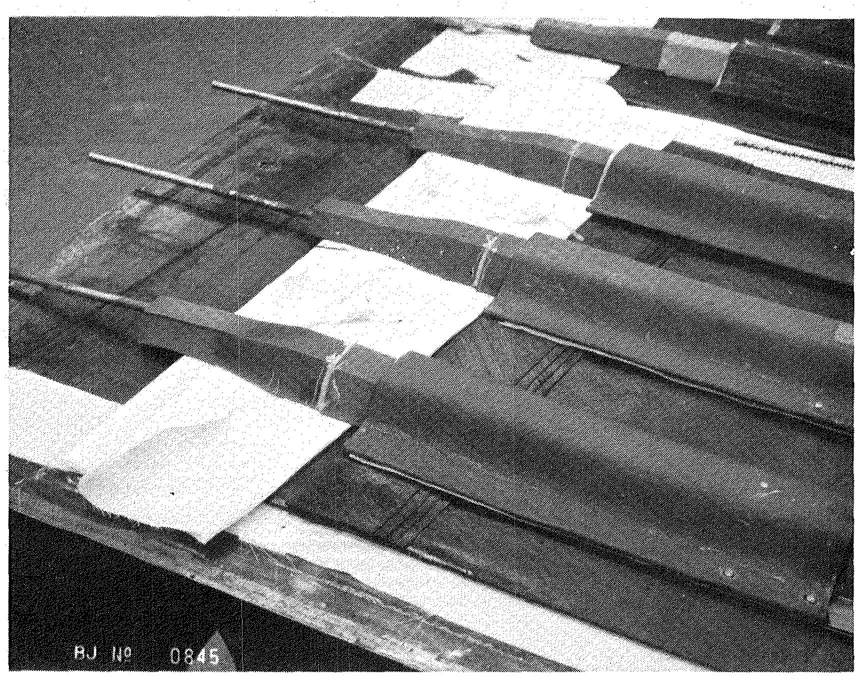



Figure 5-2. Root End Details

Figure 5-3. Layup and Hat Location Complete

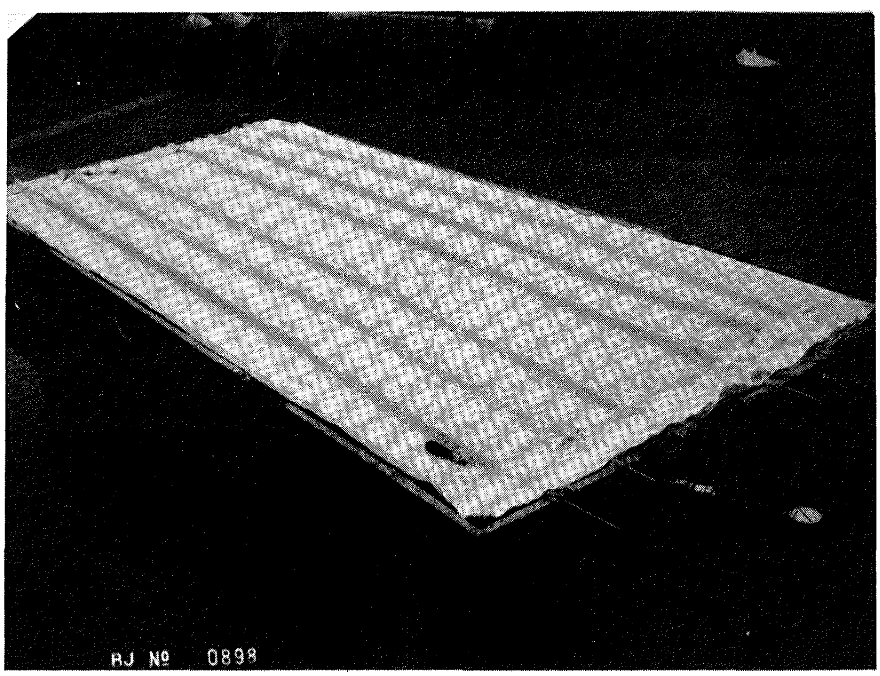



Figure 5-4. Installation of Bleed System Materials.

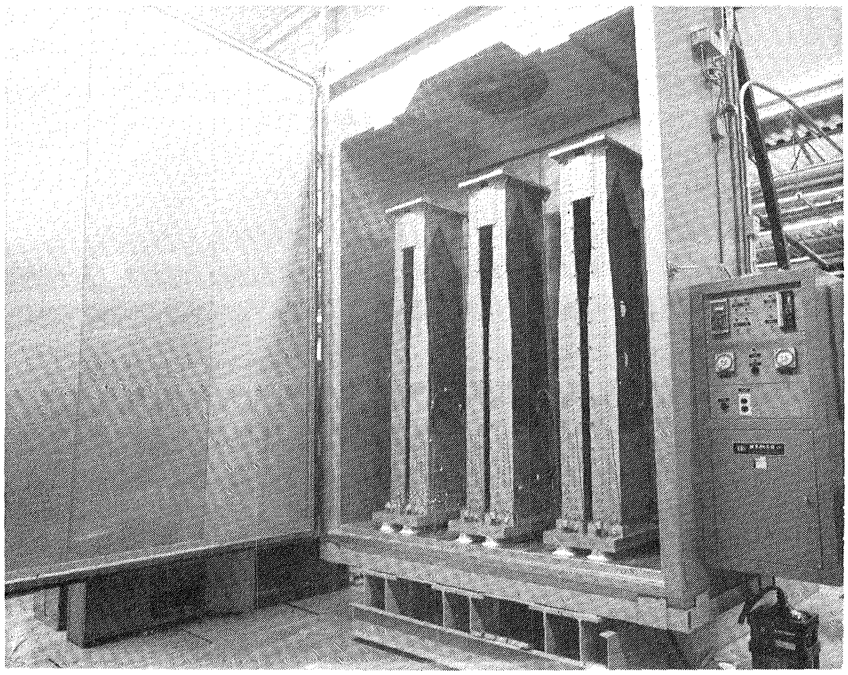


Figure 5-5. Six Spar Chamber

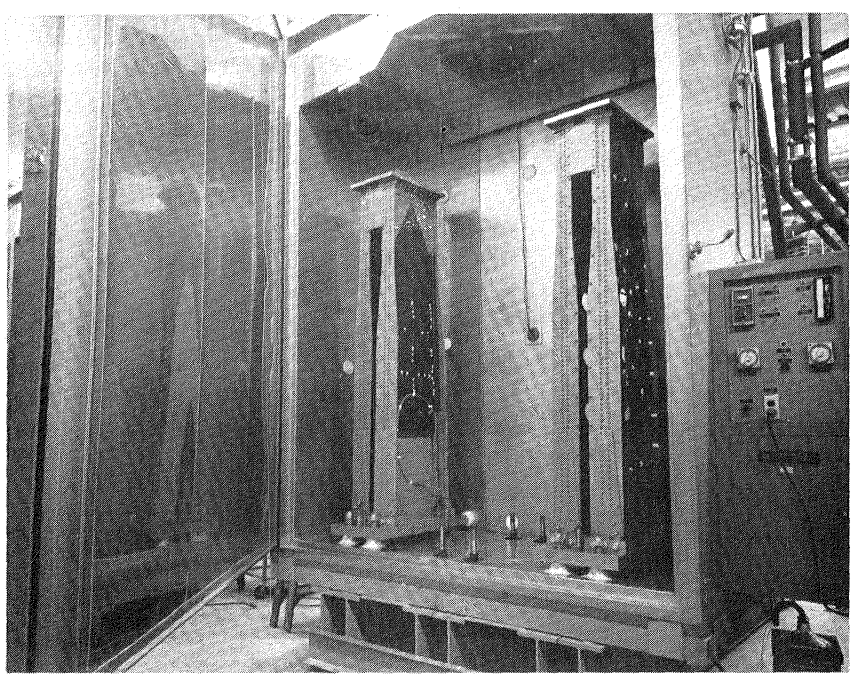


Figure 5-6. Four Spar Chamber

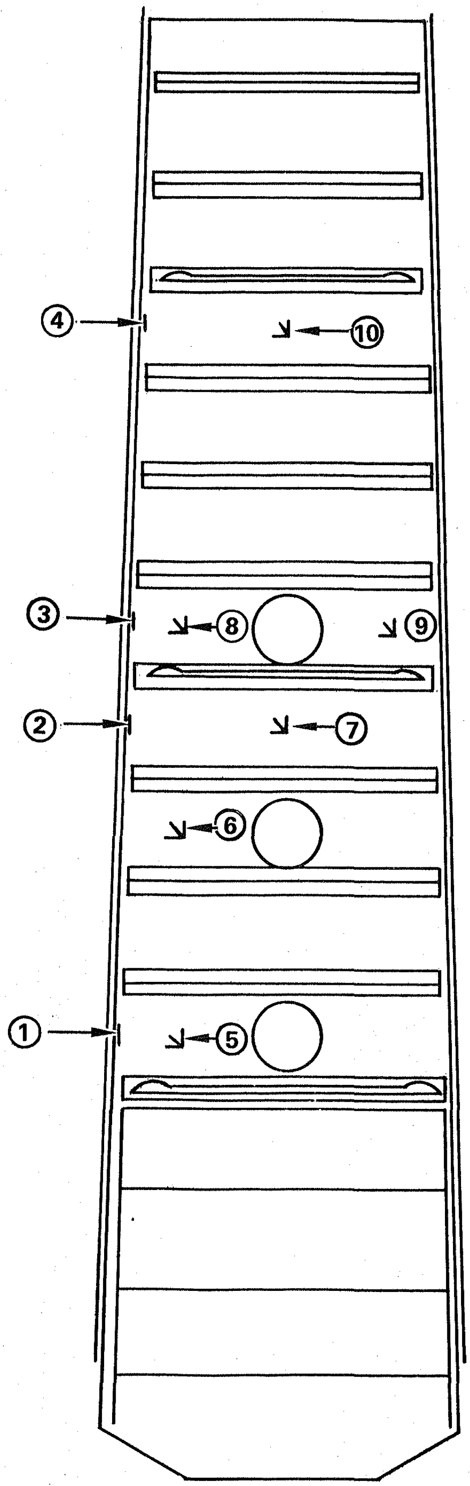



Figure 5-7. Spar Strain Gage Locations

underway. Restraint fixtures have been installed outboard of the load cell/ jack to allow spectrum flight loads to be applied to the load cell and the signal fed back to the computer for a closed-loop system check. Checks to date on both spar servo loops verify load calibration and system control. The mechanical jack stops have also been verified at full system pressure.

Checkout of the chamber environmental control system is in progress. Limited temperature control of each chamber from their individual controllers has been demonstrated and thermal mapping and baffling are in progress. In the four spar chamber a total of thirty-five thermocouples have been installed for mapping. These include locations inside the box structure in back-to-back locations for possible gradient measurements. In Figure 5-6 part of the thermocouples can be seen in the center of the open graphite specimen area held down by white RTV rubber adhesive patches along the wires in approximately one inch intervals. Additional permanent thermocouples are mounted in the center of each chamber pair and on the outlet, to monitor left and right heater bank performance and balance as an added safety precaution.

## 5.3 SPAR STATIC

All ten spar static test specimens have been completely assembled. Strain gages have been applied to all specimens. All spars have the full set of four axial and six rosette gages as shown in Figure 5-7. One spar has been installed in a test fixture and is mechanically complete as shown in Figure 5-8. Testing to failure will occur during April 1979. In addition to the strain gages, deflection will be measured at five places as shown in Figure 5-9. Testing will include loading to limit and ultimate several times and then to failure. Data will be recorded by the Rye Canyon Central Data computer system, which will allow quicklooks of the data on a CRT terminal and subsequently analyze and print test results, both digitally and graphically. These results will then be compared to theoretical predictions and results of the first spar static test at Lockheed-Georgia before the other nine tests are conducted.

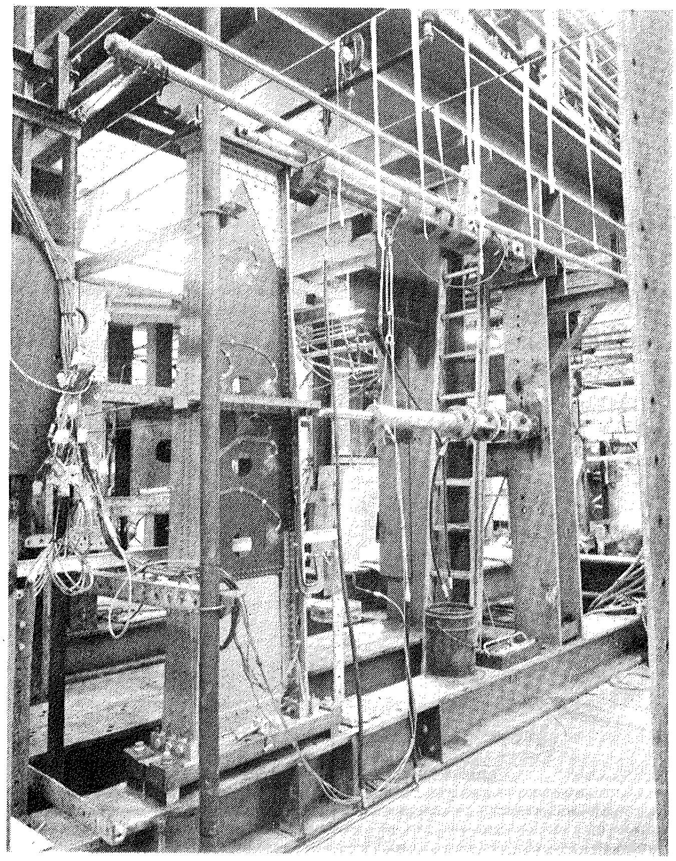


Figure 5-8. Spar Static Test Setup

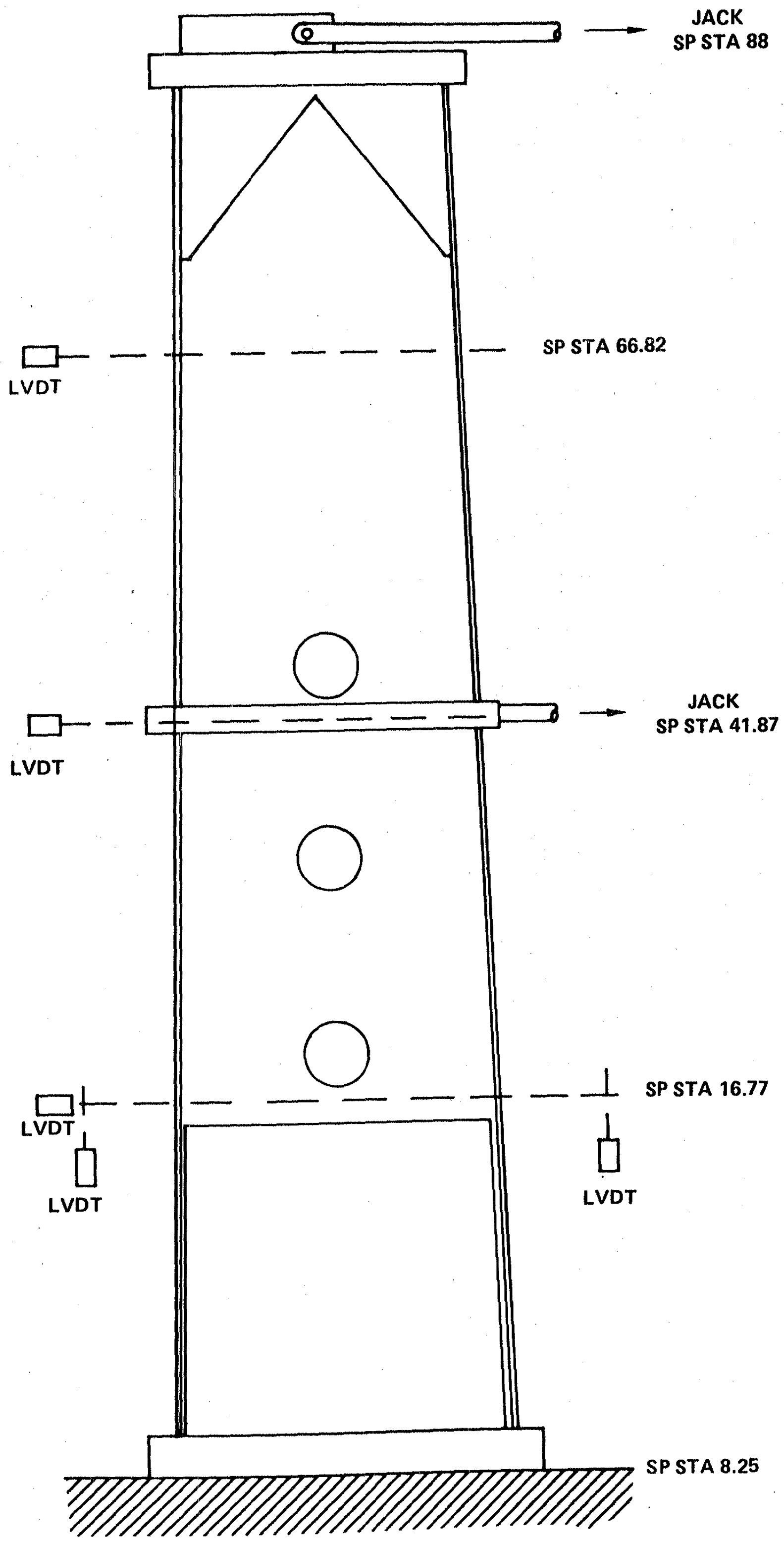



Figure 5-9. Spar Static Test Schematic

## 5.4 COVER DURABILITY

Four cover durability test specimens have been completely assembled and instrumented and are ready for installation in the test frame. A complete assembly of two covers with the buckling stabilization structure is shown in Figure 5-10. In Figure 5-11 one side of the stabilization structure has been removed to expose details of the internal flexure restraint design.

Seven axial and one rosette strain gages have been applied to one cover of the first four, the remaining three have only one axial strain gage (#2). As in the spar durability test it is planned to have one fully instrumented specimen in each environmental chamber with reduced instrumentation on all others to keep within available data monitoring channels. Gage location is shown in Figure 5-12. Gage #3 is for static testing only and will not be used for durability testing.

It is planned to install the first six covers in one chamber and start the durability test as quickly as that chamber can be brought on line. The fifth and sixth durability covers are currently being assembled and will be completed by the time the chamber checkout is complete.

## 5.5 COVER STATIC

Six cover panels have been received for test preparation. Two are currently being assembled and strain gages are being applied. Left-hand test fixtures have been fabricated and the right-hand H-25 test hardware refurbished.

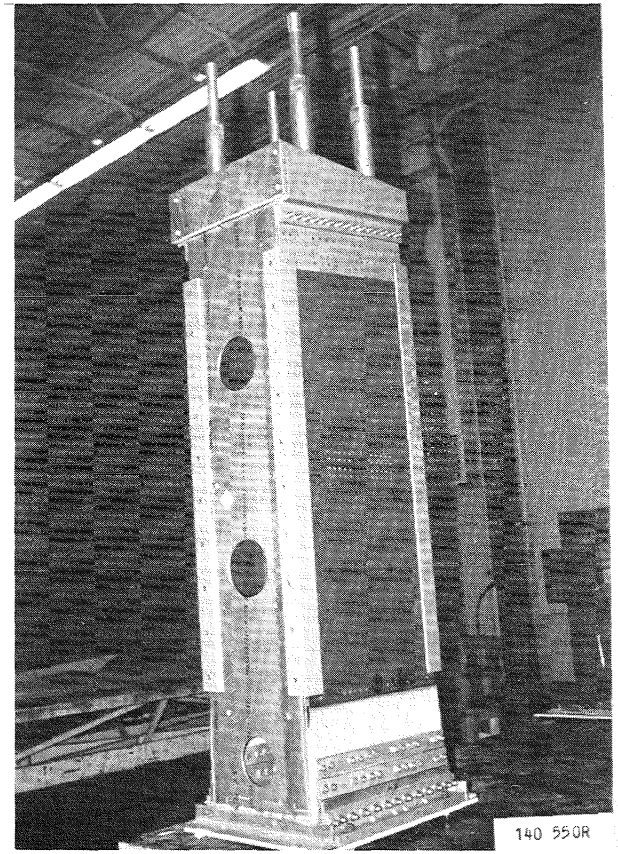



Figure 5-10. Cover Pair Assembly

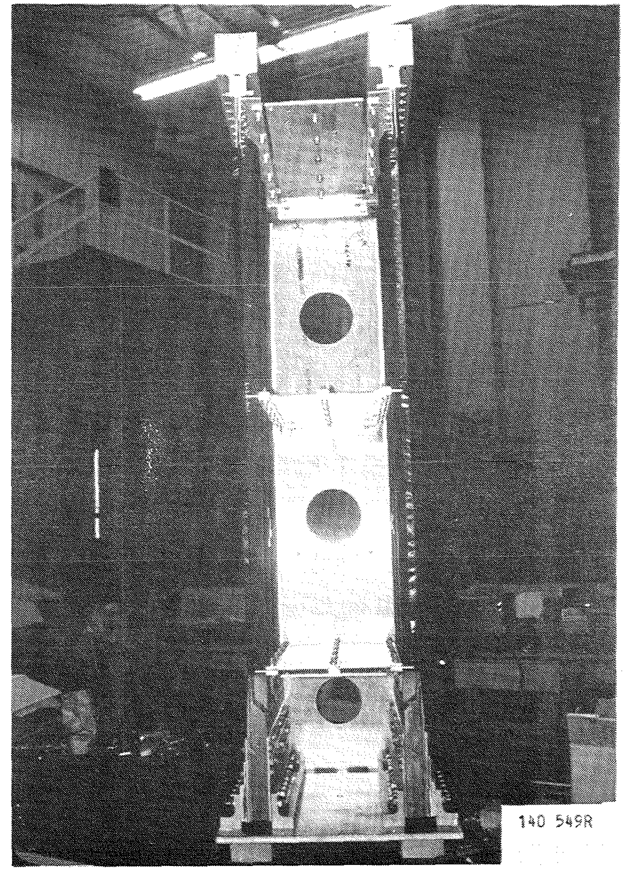



Figure 5-11. View of Inner Flexure Arrangement

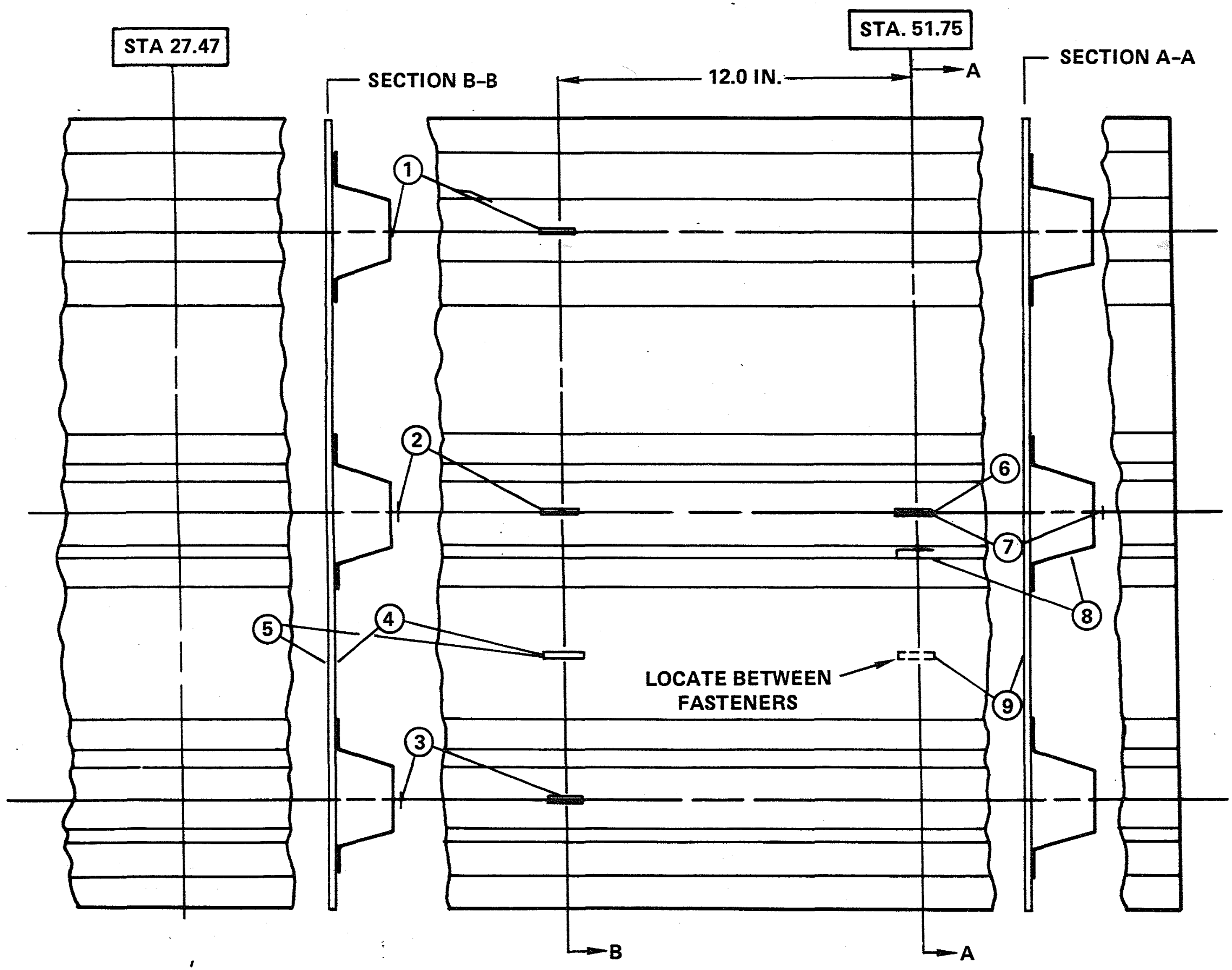



Figure 5-12. Cover Strain Gage Locations

# SECTION 6

# COVER AND SPAR FABRICATION

Two full-scale covers, one left and one right and two full-scale rear spars were fabricated during this reporting period.

## 6.1 FULL SCALE COVER FABRICATION

The first two full-scale ACVF cover assemblies have been fabricated, one left hand and one right hand. These were preceded by a left hand tool try part representing the forward two-thirds of the full-scale cover. It was approximately six feet wide at the root and had ten stiffeners including one twenty-six feet long. This section of the cover was selected for tool try as it represented the greatest variety of tool design configurations.

All of the full-scale cover components were found to present minimal problems associated with scale-up. Removal of the inflatable bladders, for example, from the hat-stiffener cavities was considered a critical feature of the scale-up effect. Bladder removal from all three assemblies has been performed without any difficulty.

The cure cycle was modified in the third assembly to comply with a mandatory safety requirement to maintain the oxygen content of the autoclave below a specific level for safe operating conditions. To meet this requirement heat and pressure were applied immediately, and when the autoclave reached 50 psi and 150°F, the oxygen content dropped to a safe operating condition. Part temperature, in the meantime, had only risen to 110°F maximum. A pressure leak check was made at this time. Pressure was reduced to 20 psi and maintained while part temperature was raised to 210°F. After dwell at 210°F was complete, part temperature was raised to 260°F. After dwell at 260°F was complete, pressure was raised an additional 65 psi to comply with minimum pressure of 85 psi. The remainder of the cure cycle followed the nominal requirements of the process bulletin.

The bleeder stacking used on the right hand cover assembly was also modified to reduce manhours involved in this phase of the layup operation. All bleeder plies of Nexus and 120 cloth, normally used over the skin area between hat stiffeners, was removed. The one ply strips of peel material normally used over the hat stiffeners was replaced by two plies of peel material. Since these strips also overlap each other in the skin area between hat stiffeners the peel material between hats is doubled from 2 plies to 4 plies. These additional layers of peel material provide bleed characteristics equivalent to the 120 cloth and the Nexus it replaces and results in a reduction in manhours for layup. The additional ply of peel material over the hat stiffener is expected to improve the crown thickness per ply in the laminate. The barrier film normally applied in strips between the hat stiffeners and the underside of the caul was replaced by a continuous sheet of barrier film allowed to drape between hat stiffeners.

A sequence of photographs is provided in the following figures to illustrate some of the features of full-scale fabrication.

Figure 6-1 shows front spar skin doubler plies being laid in position. Figure 6-2 shows the tape dispensing machine used for locating $0^\circ$. Figure 6-3 shows backing paper being removed from $45^\circ$ plies. Figure 6-4 shows a skin layup being bagged as a debulking operation at the end of the work shift. Figure 6-5 shows the first five-ply stack of preplied hat stiffener material prior to preforming on the layup block. In Figure 6-6 the $0^\circ$ plies are being located over the first five-ply stack. Figure 6-7 shows hat layup complete, with caul in place, and ready for vacuum preforming. In Figure 6-8 the hat stiffener two-ply reinforcement strips are being inserted between the hat and the inflatable bladder. Figure 6-9 shows hat stiffeners being positioned on the skin. Figure 6-10 shows three hat stiffeners being located by spacers indexed to the MBF. Figure 6-11 shows all hat stiffeners located and bladders connected. In Figure 6-12 the layup is complete and edge dams and bleeder material is being installed. Figure 6-13 shows peel ply material over the hat stiffeners and overlapping between bags. Figure 6-14 shows caul plates being reinstalled over the hat stiffeners and bleeder materials. In Figure 6-15 the completed layup is bagged and ready for cure.

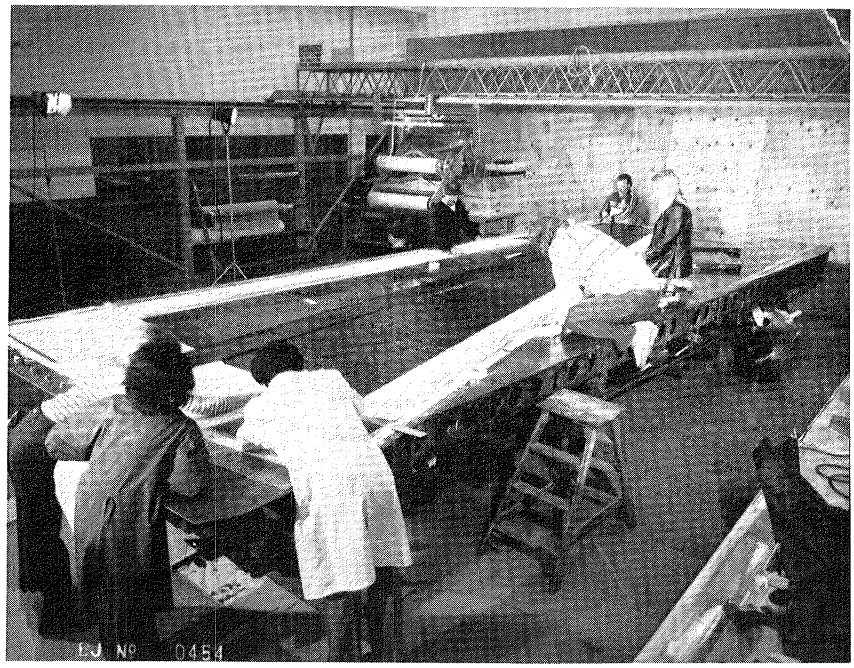



Figure 6-1. Front Spar Doubler Installation

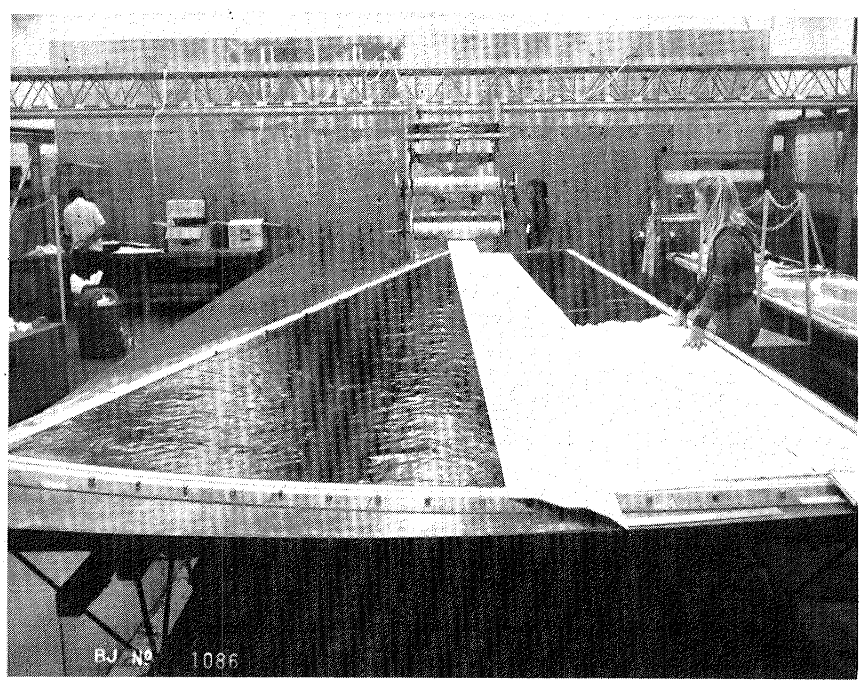



Figure 6-2. Locating $0^{o}$ Oriented Tape

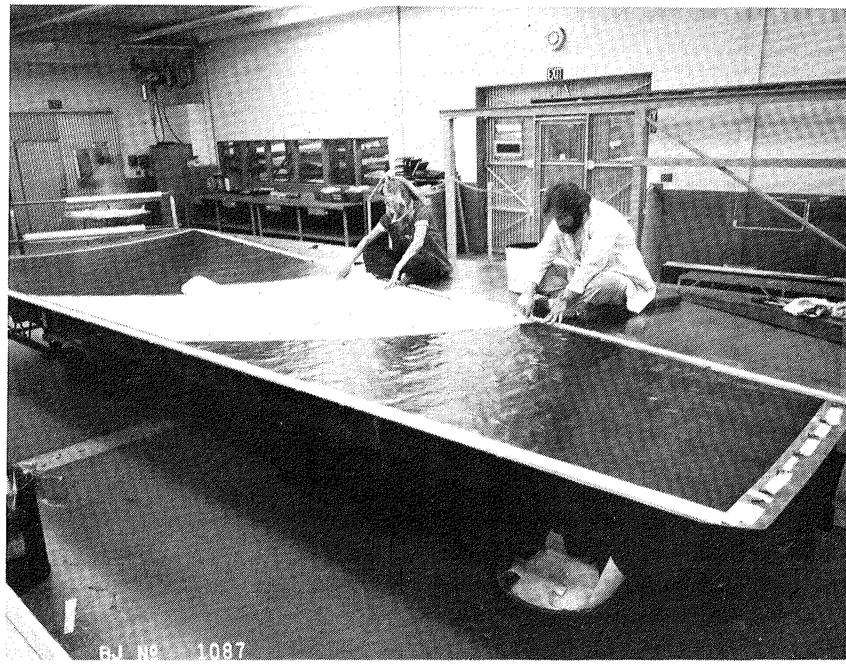



Figure 6-3. Removing Backing From 45° Plies

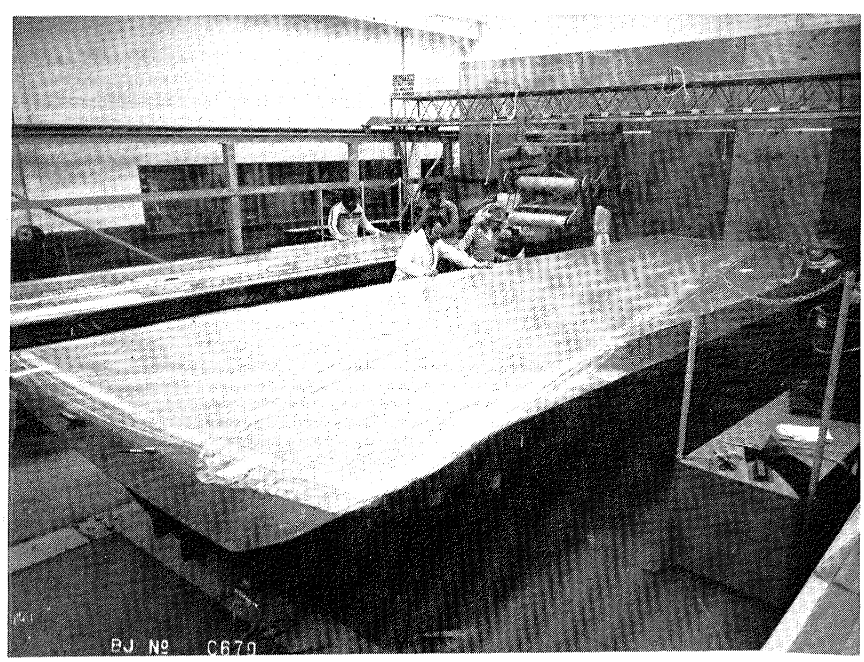



Figure 6-4. Debulking Skin Layup

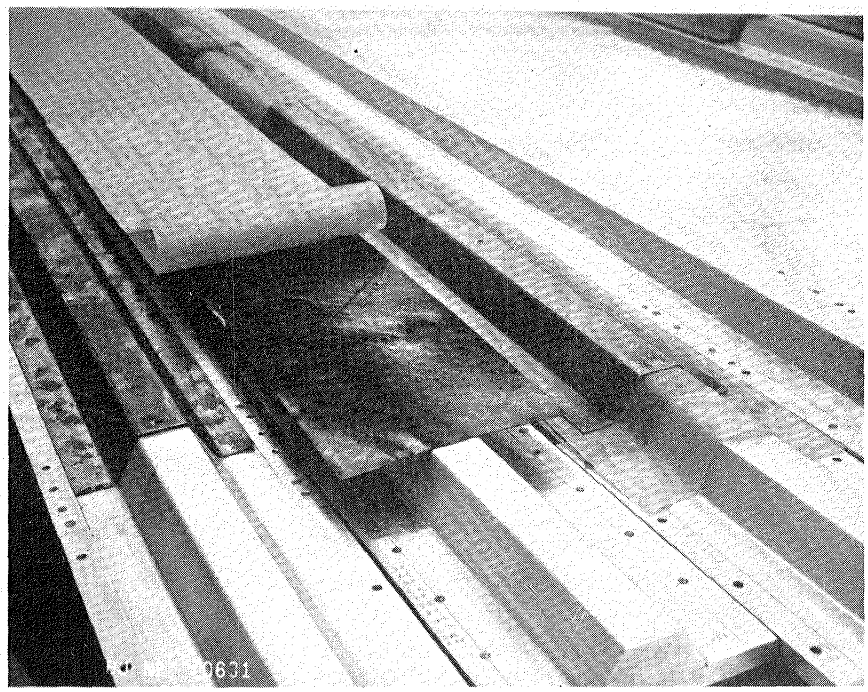


Figure 6-5. Preforming Hat Stiffeners on Layup Block

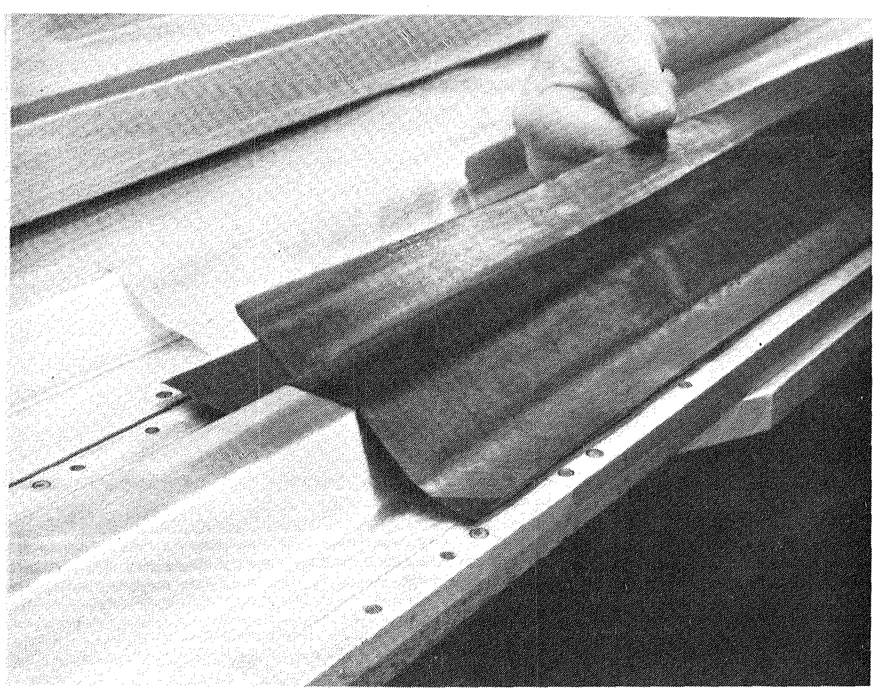


Figure 6-6. Locating 10 Ply $0^{o}$ Stack on Crown

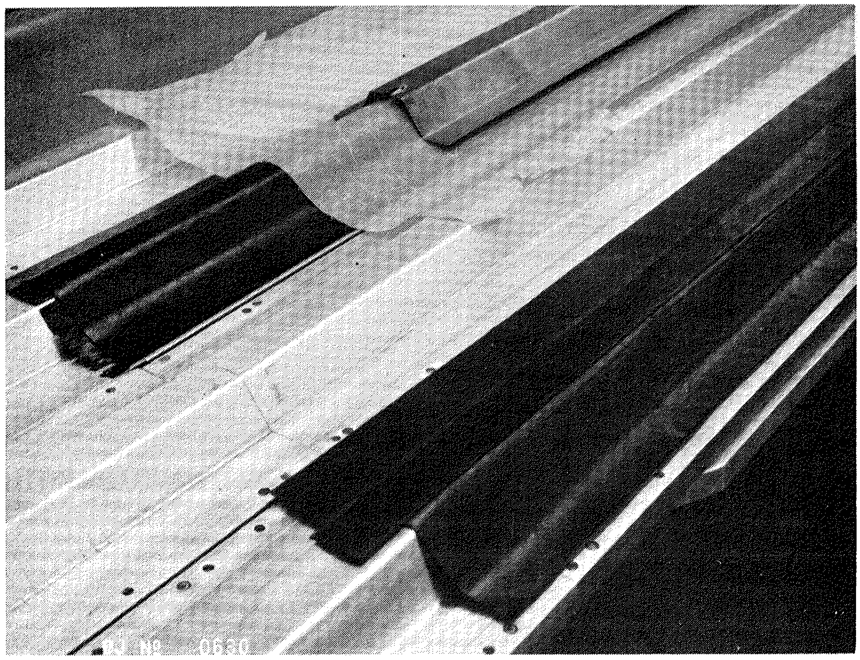



Figure 6-7. Hat Stiffener Ready for Final Preforming


Figure 6-8. Locating Hat Stiffener Reinforcing Clips

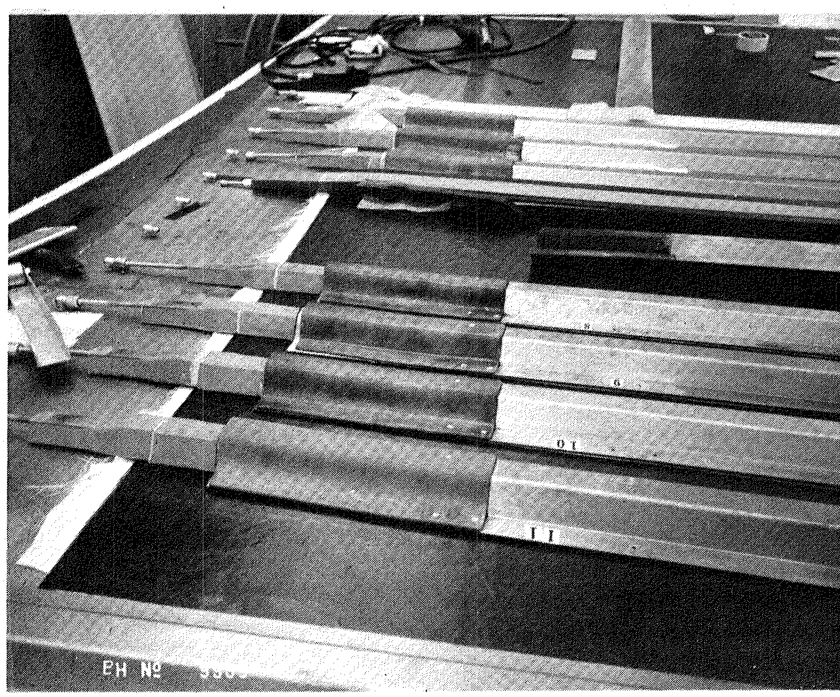


Figure 6-9. Hat Stiffeners Positioned On Skin

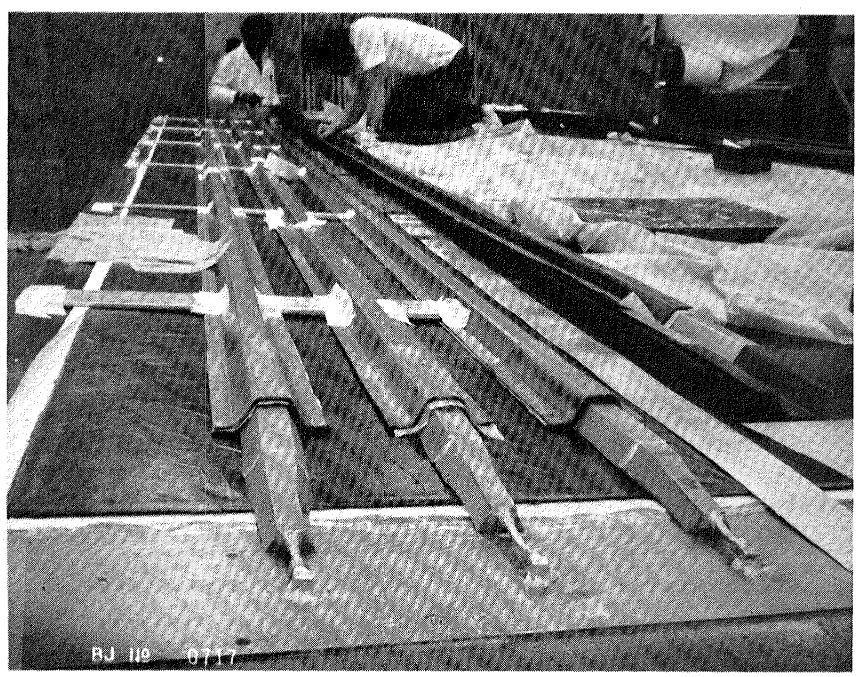


Figure 6-10. Hat Stiffener Locators Being Installed

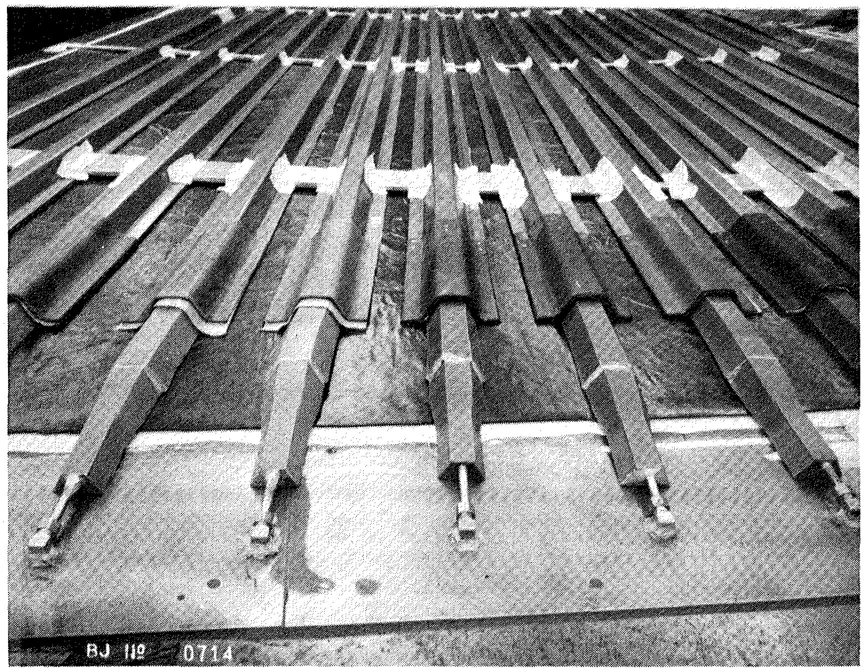


Figure 6-11. Complete Assembly, All Locators Installed

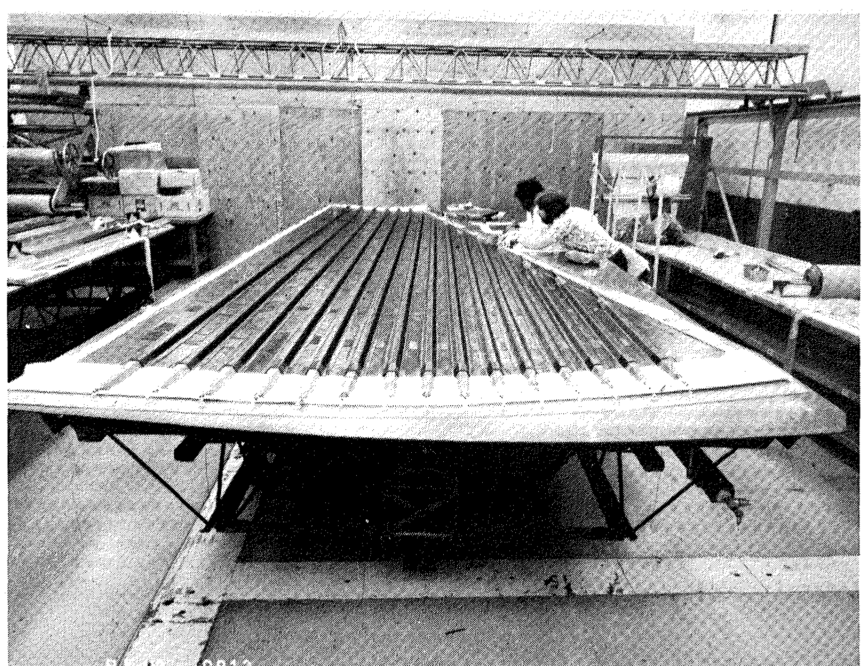


Figure 6-12. Edge Dams and Bleeder Being Installed

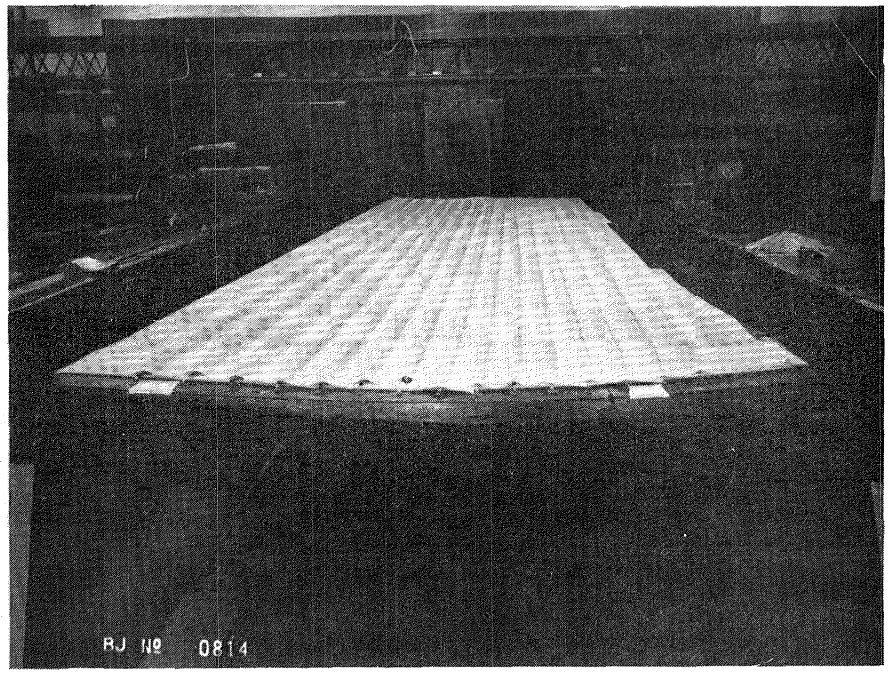



Figure 6-13. Peel Ply Installed Over Hat Stiffeners

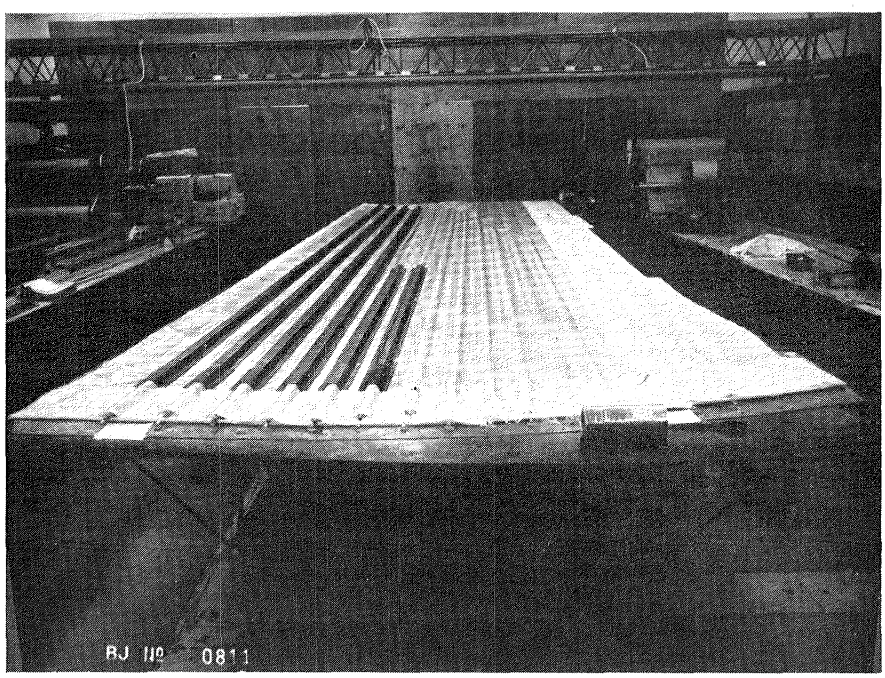



Figure 6-14. Caul Plates Being Installed Over Hat Stiffeners

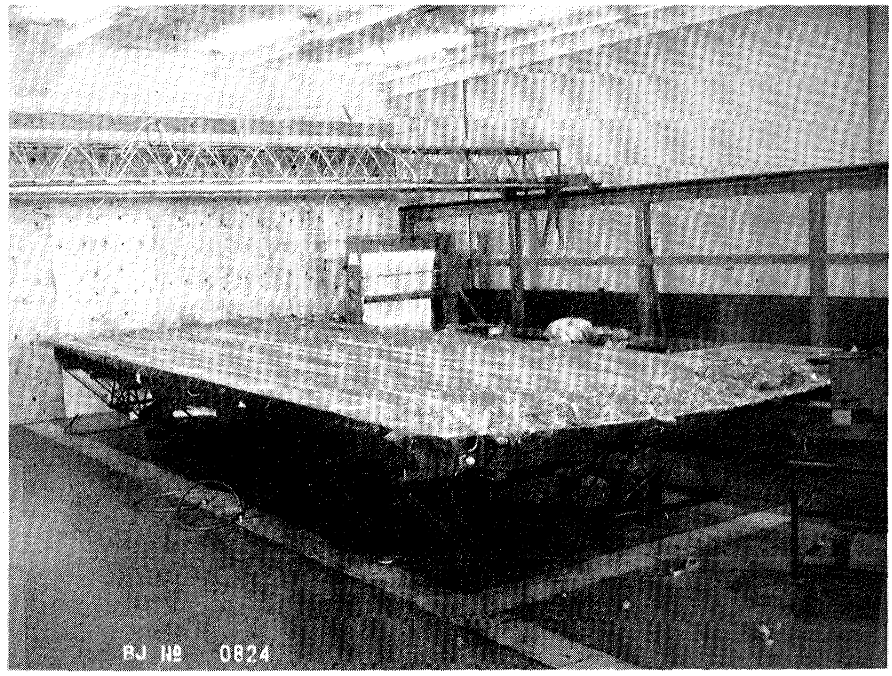



Figure 6-15. Assembly Bagged and Ready for Cure

In Figure 6-16 the bag has been removed from the cured assembly and shows resin pickup in the breather material. Figure 6-17 shows the skin side of the cured assembly after the peel ply material has been removed. Figure 6-18 shows the cured assembly hat side up. Figure 6-19 shows cured assembly mounted in a picture frame tool ready for ultrasonic inspection. Figure 6-20 shows water nozzle set-up for through transmission ultrasonic inspection.

A silicone rubber sheet is being evaluated in lieu of the conventional nylon vacuum bag for the purpose of eliminating pin holes, bag failures at temperature, and bag ruptures at pressure due to bridging. The concept involves the use of cured silicone sheet approximately .050" thick, draped over the assembly, and sealed to the MBF by clamps mounted to pressure bars. Figure 6-21 shows the silicone bag draped over hat stiffener caul plates and clamped around the periphery. Figure 6-22 shows the same set up after vacuum was applied. The bag sucked down very uniformly over the hat

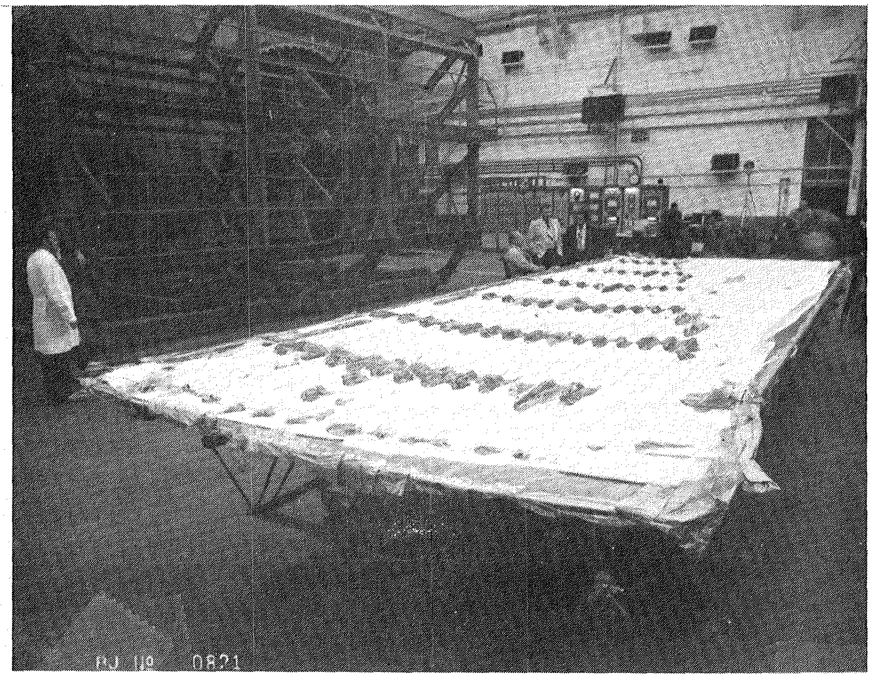



Figure 6-16. Cured Assembly With Bag Removed

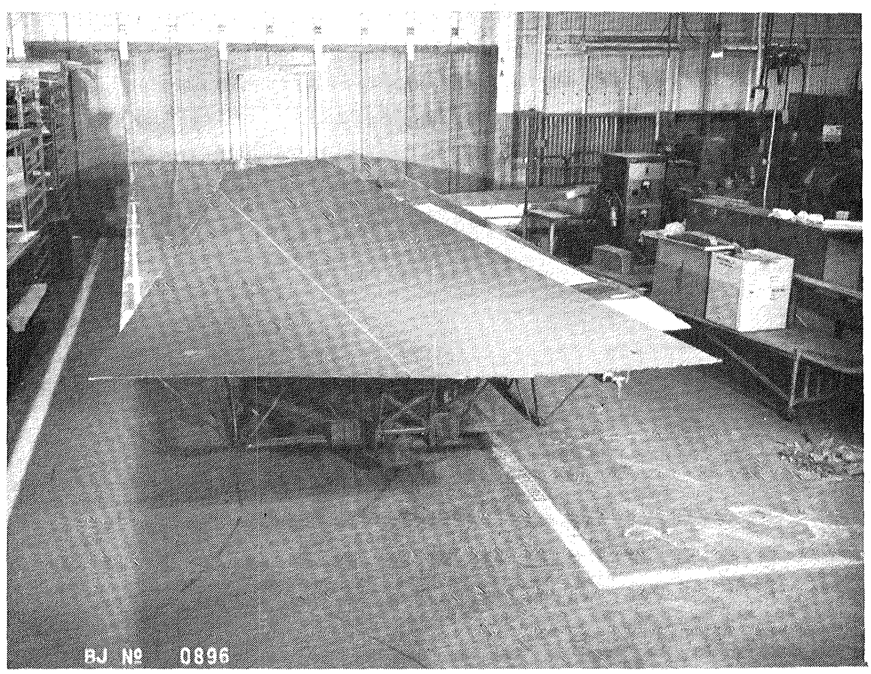



Figure 6-17. Cured Assembly Skin Side

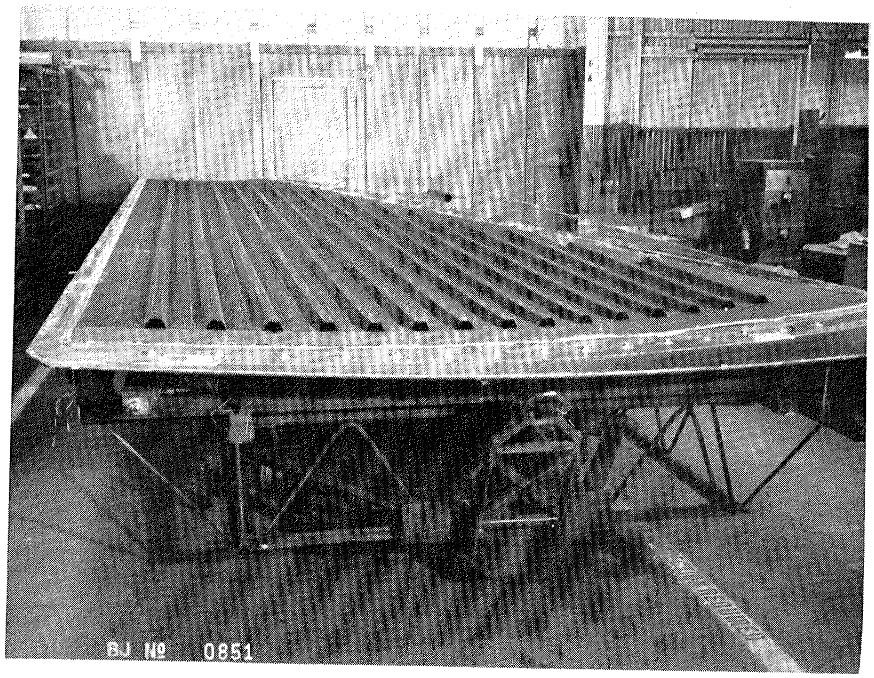



Figure 6-18. Cured Assembly Hat Side

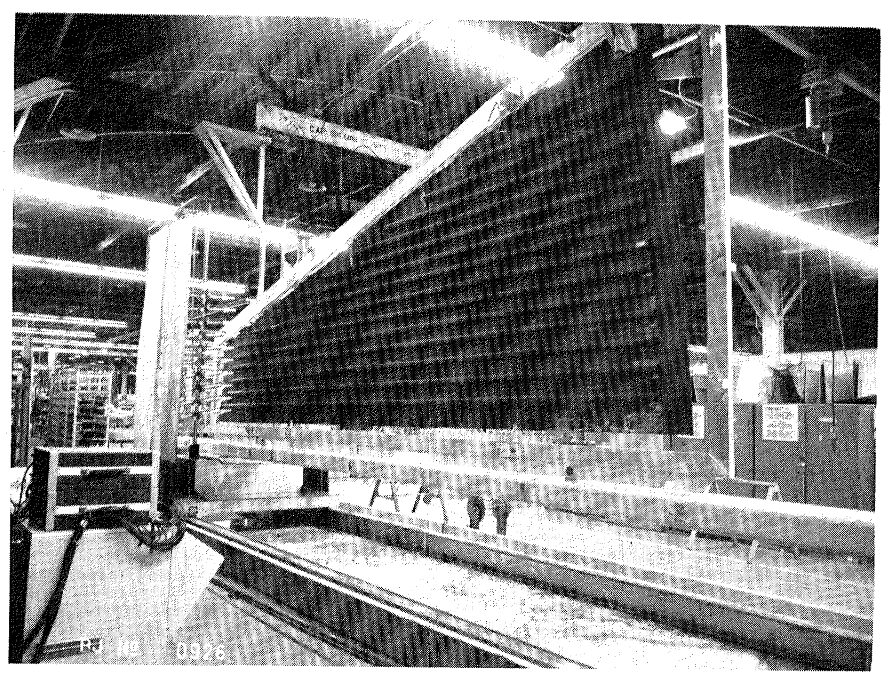



Figure 6-19. Cured Assembly Ready for Ultrasonic Inspection

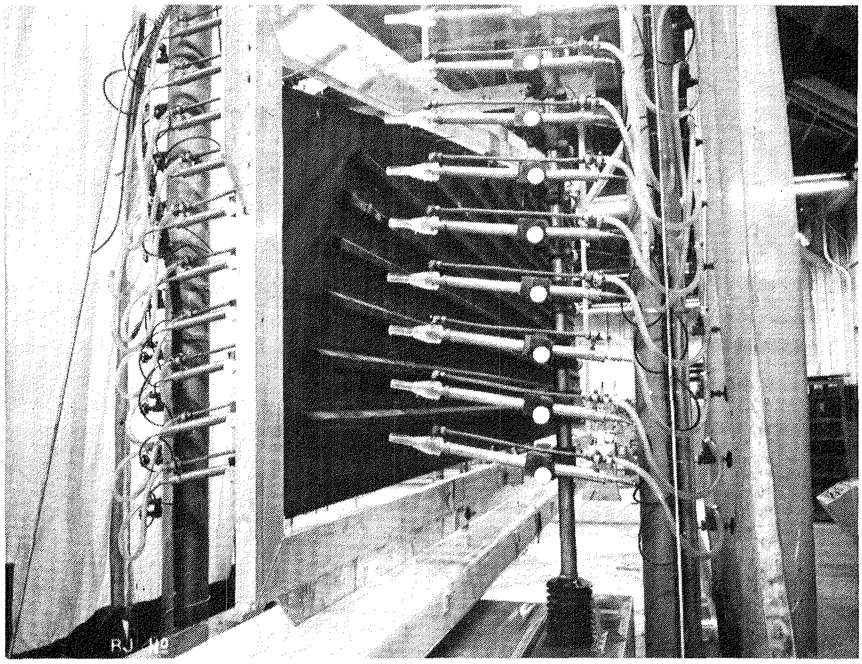


Figure 6-20. Ultrasonic Water Nozzle Setup

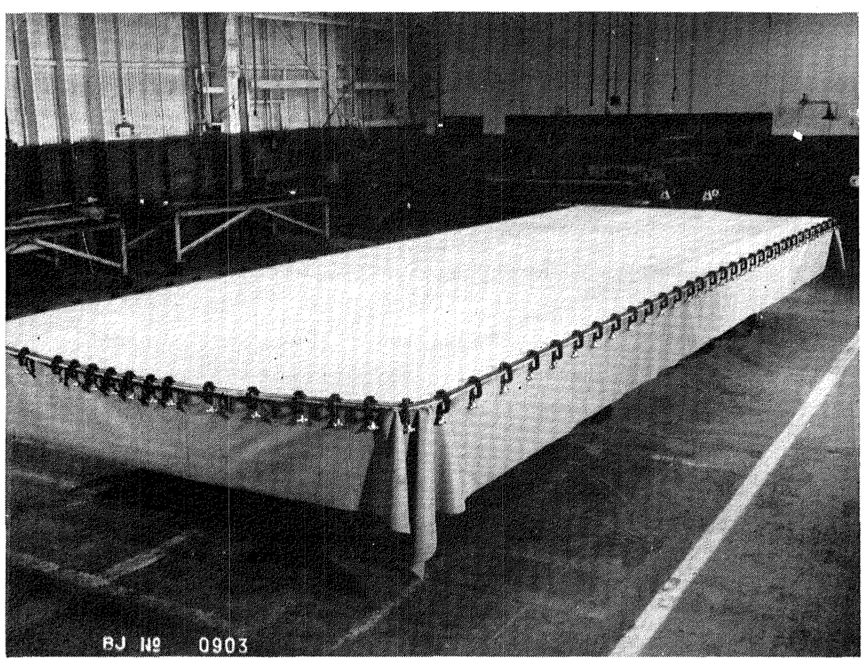



Figure 6-21. Silicone Bag Prior to Application of Vacuum

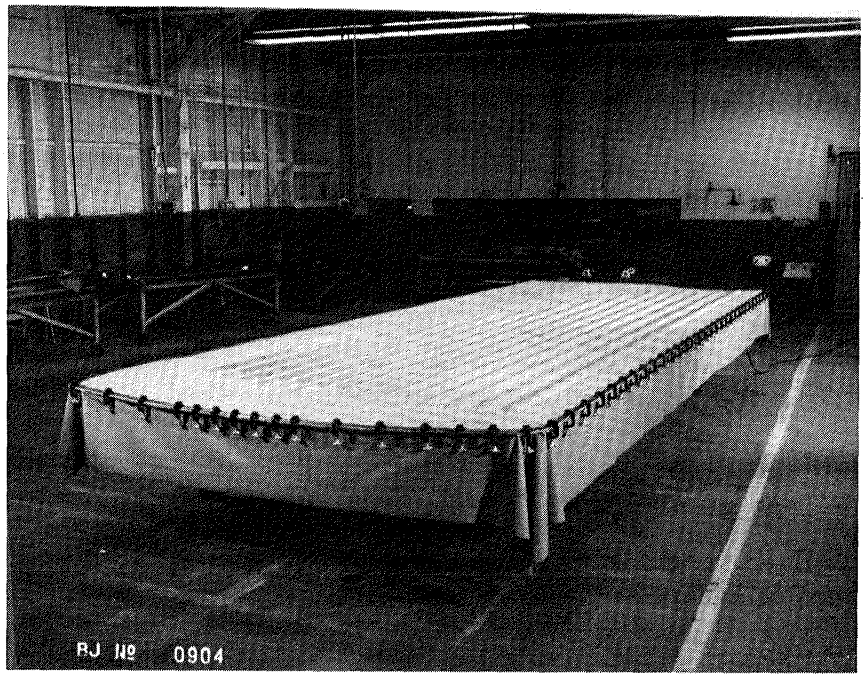


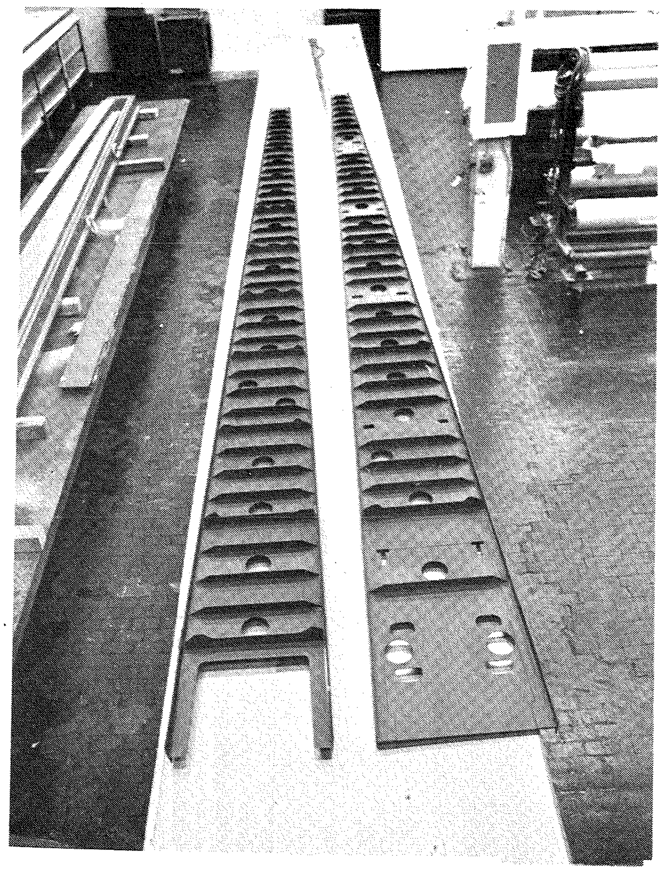

Figure 6-22. Silicone Bag After Vacuum Applied

stiffener caul plates. There was no evidence of bridging. However, when vacuum was shut off there was a vacuum loss. Tracing the source of the leakage is in progress and if the concept is proven, it is expected to yield both cost and quality improvement.

## 6.2 FULL-SCALE SPAR FABRICATION

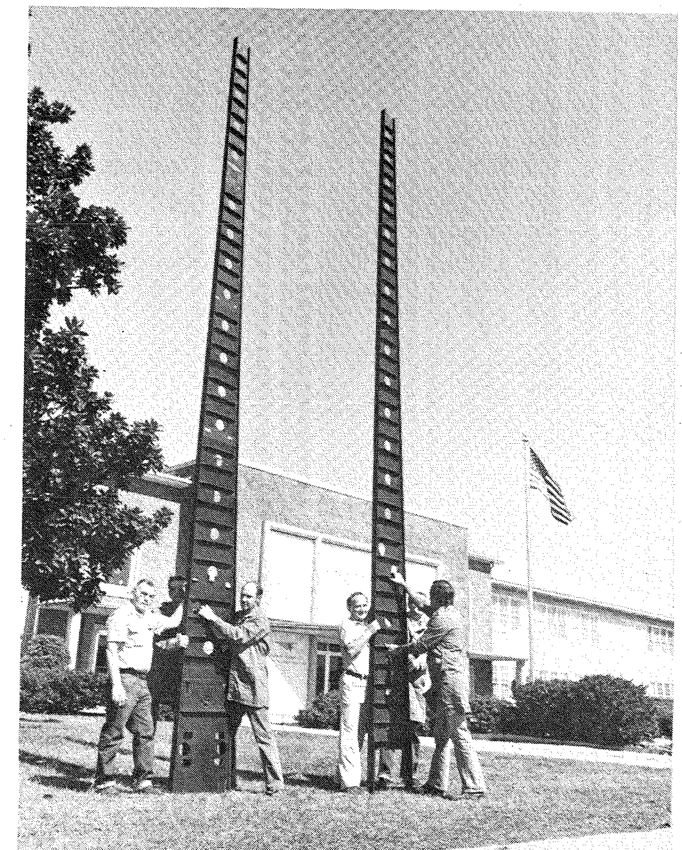
One full-scale front spar and two full-scale rear spars have been completed. The third rear spar has been laid up in kit form. Figures 6-23 and 6-24 show the first ship set of spars.

Ultrasonic inspection was completed on the Number 1 and Number 2 rear spars, and no porosity or voids were found. The dimensional inspection of Number 1 was completed, and it generally conformed with the specified tolerances except in the aft flange of the spar cap where thickness was slightly over tolerance. The access holes and tag ends were removed from the spars and cut into process control specimens for mechanical and physical tests. Figure 6-25 illustrates the procedure used to remove the access holes.



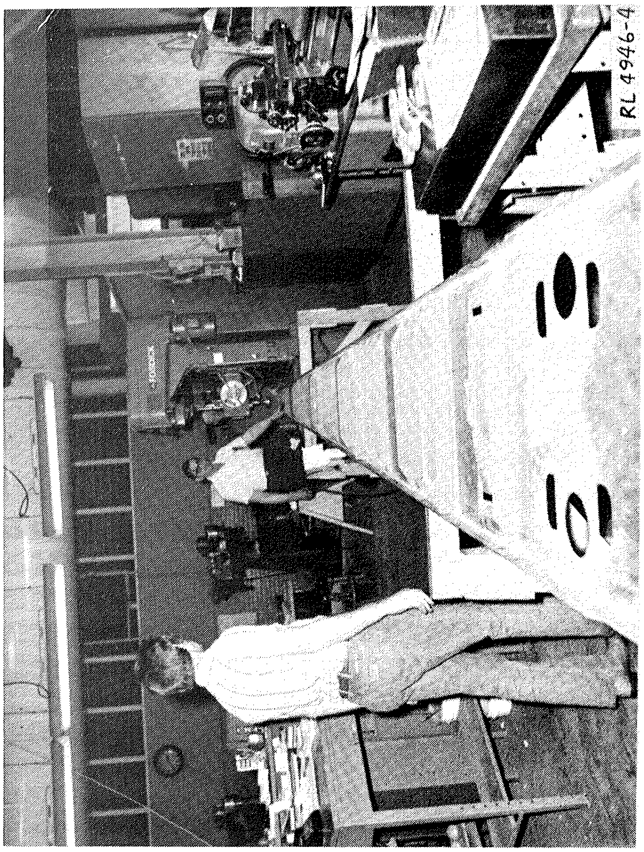
RL 4953-1

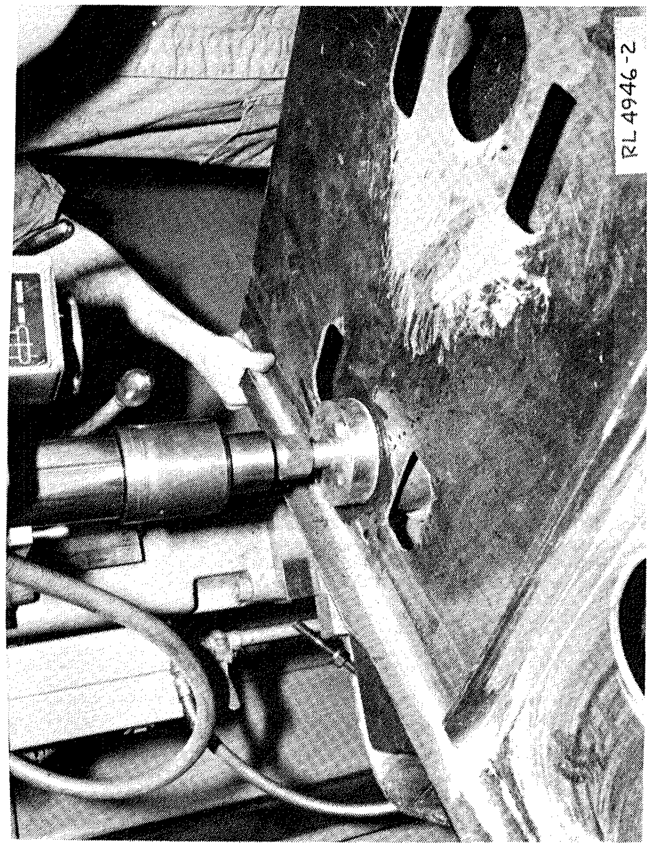
Figure 6-23. First Ship Set of Spars

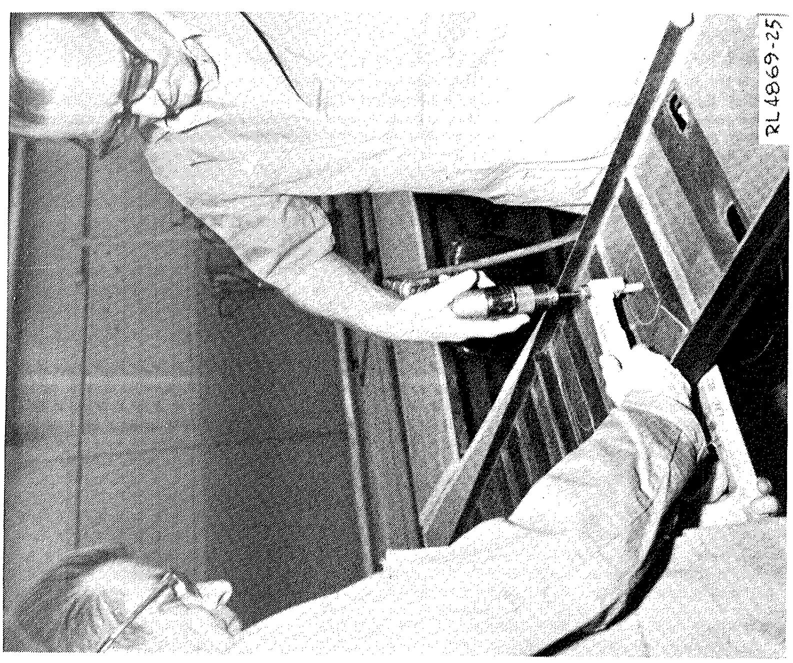


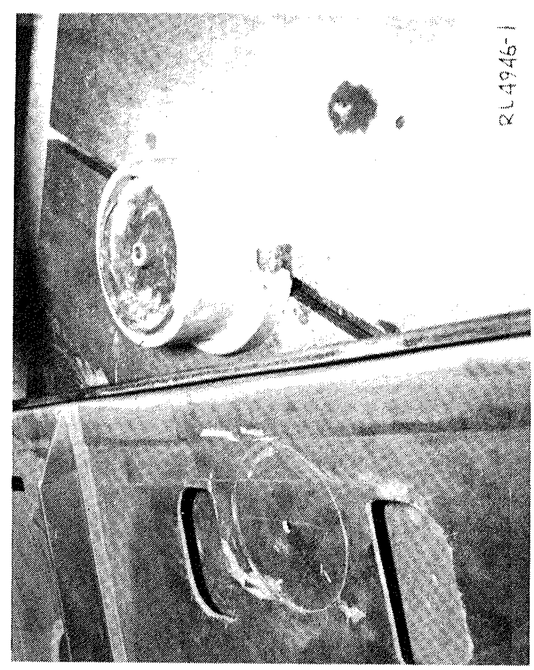
RL 4953-2

Figure 6-24. Rear Spar and Front Spar

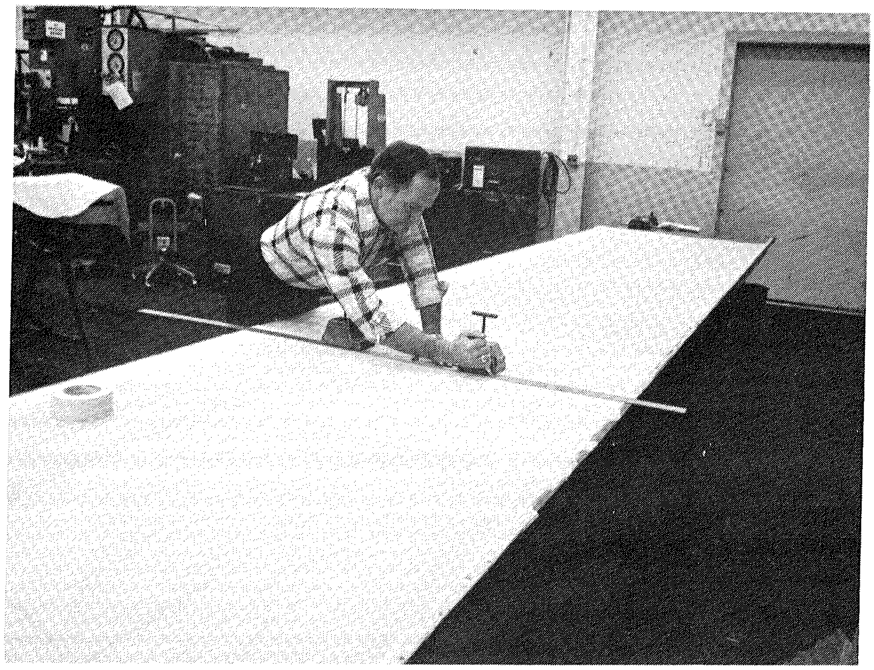

RL 4946-4


RL 4946-2


RL 4869-25


RL 4946-1

A tooling hole is drilled to locate the 4-inch diameter disc cutter and the disc is cut out of the access hole as shown in the figure. The disc removed from the 4-inch diameter access hole is 3.75 inches in diameter, and it is used to obtain compression, short beam shear, flexural, resin content, specific gravity and porosity process control specimens. The third rear spar was laid up, kitted and placed in the freezer. The 12-inch wide tape is used to lay out a wide sheet of "broad-goods" approximately 36 inches by 30 feet with four to six plies. A computerized lay out is used to "nest" the ply patterns in the most efficient arrangement to minimize waste. The full size pattern is placed on the broad-goods sheet and the individual elements are cut out as shown in Figures 6-26 and 6-27. Although hand cutting is shown, the set up is ideal for future adaptation to use of a numerically controlled cutter.



RL 5011-12

Figure 6-26. Full Size Pattern

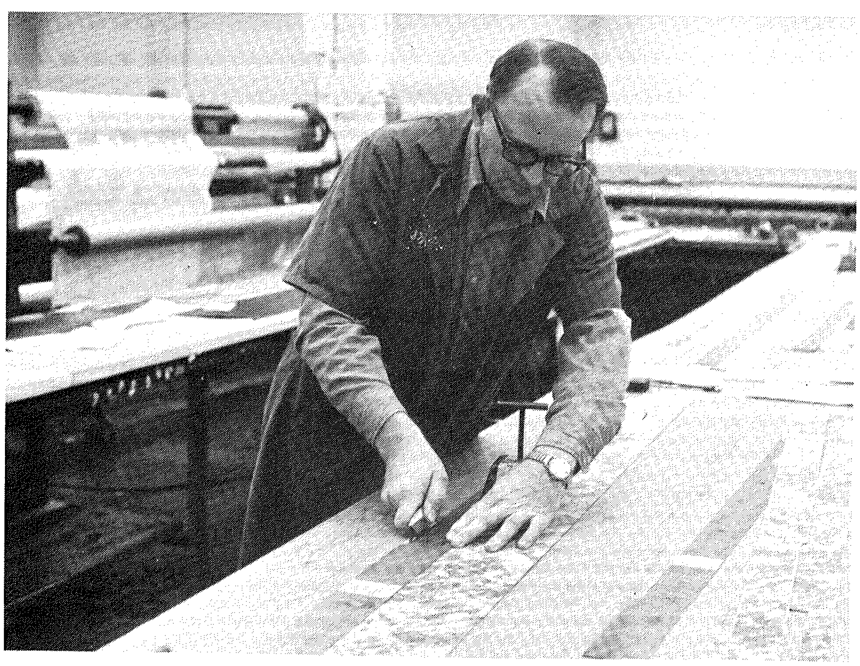


Figure 6-27. Cutting Elements to be Used in Spar Assembly from Wide Laminate

# SECTION 7

## CAUSE AND EFFECT OF WRINKLES IN PREBLED PANELS

In the development of the fabrication process for cocuring the hat stiffened skin panels utilizing high resin content prepreg material, it was necessary to prebleed in order to minimize the amount of resin bleeding during the cure cycle. It was observed that during the prebleeding operation wrinkles were being formed after cool-down from the 225 F prebleed temperature. The wrinkles were of concern because if they were present after final cure, indicating wrinkled (kinked) fiber, the structural integrity of the laminate (component) might be severely limited. Therefore, steps were taken to identify the cause of wrinkles in prebled panels and to instigate a remedy to eliminate them. Concurrent with the wrinkling study, cured laminates which had wrinkles formed during the prebleeding stage were examined to determine if in fact they were present in the final cure. A typical wrinkled panel prior to final cure is shown in Figure 7-1. The panel was 100 inches long and 48 inches wide and consisted of six plies of graphite/epoxy prepreg material.

### 7.1 STUDY OF CAUSES OF WRINKLING

The observed facts are as follows:

1. The wrinkles are being formed during the cool-down from the 225°F prebled temperature or subsequent to cool-down. There are no wrinkles while the part is above some critical temperature.

2. By physical appearance there are three kinds of wrinkles:

   (a) Sharply defined straight-line wrinkles following fiber directions

   (b) Irregularly shaped wrinkles lacking orientation

   (c) Wrinkles following the contour of doubler sections or tapered sections

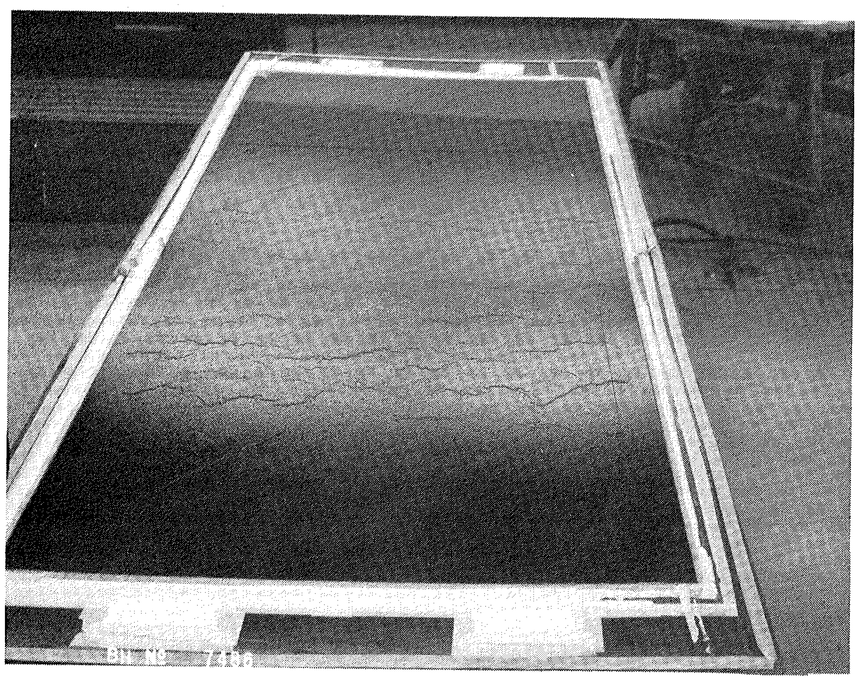


Figure 7-1. Typical Wrinkled Panel

These observations led to the following conclusions as regards the failure mechanism:

1. The wrinkles are being formed as a result of thermal stresses being generated within the stack during the cool-down process, i.e., they result from a mismatch in the thermal coefficients of expansion between the fibers, resin and bleeder materials. These thermal stresses (compressive) act upon the graphite laminate while the latter remains in the uncured state, i.e., lacks resistance toward compressive (buckling) forces particularly within the thin (5 or 6-ply) areas. Figure 7-2 shows the standard prebled layup and Table 7-1 lists material description and supplier.

2. Contributing factors may be as follows:

   - Uneven tension within any of the bag materials
   - Asymmetrical ply structure
   - Within-ply variations in resin content (tape mating lines) or fiber tension
   - Degree of resin staging

3. The formation of wrinkles may further be influenced by process parameters affecting resin mass distribution during cool-down (cooling rate, applied pressure, resin flow restrictions).

### 7.1.1 Definitive Tests

A series of tests were defined to evaluate the above conclusions.

Possible Cause Number 1:

Resin accumulation in the polyester (Airweave SS) bleeder mat, in combination with the presence of the nylon bagging film, provided a region of high thermal contraction during cool-down, which may contribute to the formation of compressive stresses and wrinkling of the layup. To investigate the problem four (4) six-ply asymmetric 36 in. x 18 in. laminates were laid up and simultaneous prebleed was performed.

a) using Airweave bleeder (2 panels)

b) using no bleeder material

c) using 181 glass bleeder instead of Airweave

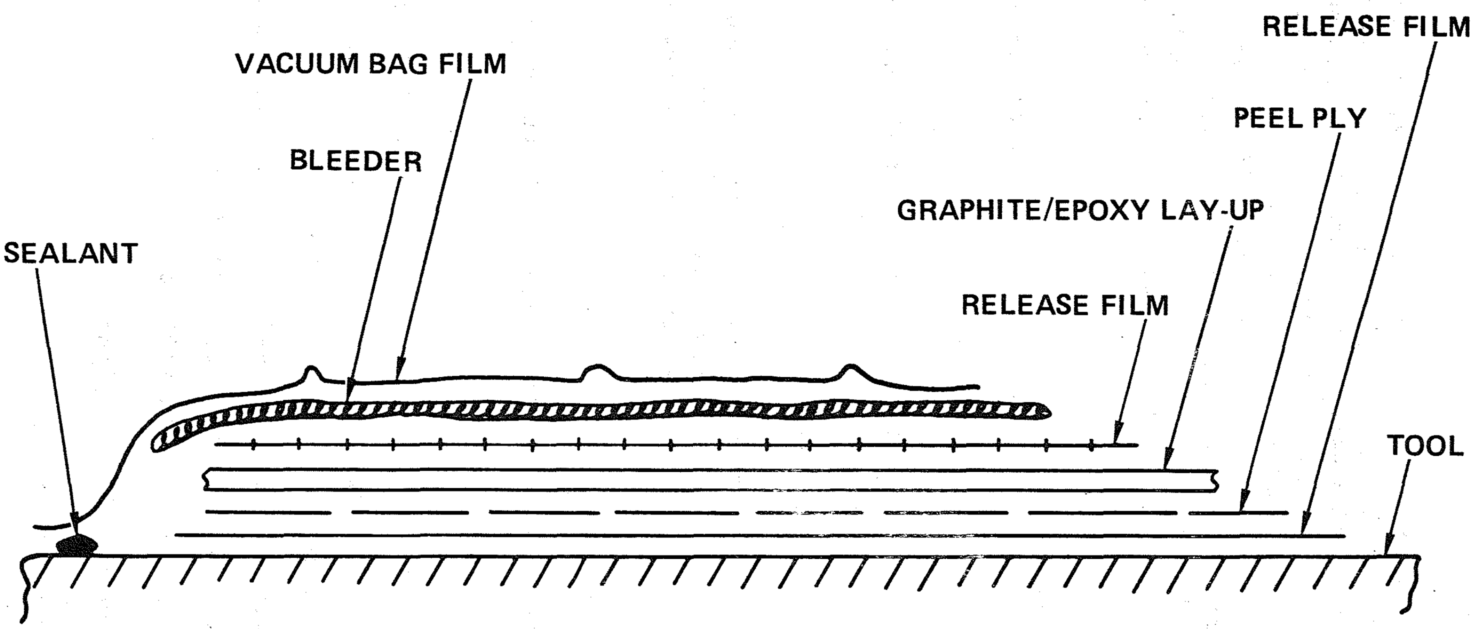



Figure 7-2. Standard Prebleed Lay-Up Construction

TABLE 7-1. AUXILIARY MATERIALS

| | | |
|---|---|---|
| <u>Standard Materials</u> | | |
| Vacuum Bag | | Wrightlon SB 7400, Int'l Plastic Prod. Corp. |
| Bleeder | | Airweave SS, Airtech, Inc. |
| Release Film, Halogen, Perforated | | A4000P Airtech, Inc. |
| Peel Ply, Nylon Taffeta | | Quality 7050 (no finish), Checkout Fabrics |
| Release Film, Halohydrocarbon, Non-perforated | | Wrightlon 4500 Int'l Plastic Prod. Corp. |
| Sealant | | GS-213 Airtech Int'l Corp. |
| <u>Alternate Materials</u> | | |
| Release Film, Halogen | | A4000 Airtech, Inc. |
| Bleeder, 181 Glass Fabric | | <u>Open Bid</u> |
| Bleeder Mats, Polyester Fiber, Hi Absorption | | Style 4819 Burlington Ind. Industrial Fab. Div. |

Use of Airweave SS (which was felt to have high difference in expansion coefficient with respect to layup) caused the most severe wrinkling; glass fib er, with less difference in expansion coefficient, resulted in less wrinkling; and elimination of bleeder also minimized wrinkling.

Possible Cause Number 2:

Shrinkage in layup-tool separator film may introduce compressive stress in layup and enhance wrinkling.

To investigate this problem three (3) 36 in. x 18 in. laminates were laid up with simultaneous prebleed as follows:

a) six-ply asymmetric with A-4000 film separator between tool and laminate

b) six-ply asymmetric without film separator

c) five-ply symmetric without film separator (for symmetric vs. asymmetric evaluation)

For the asymmetric layup there was no wrinkling with or without the film. However the symmetric layup without the film resulted in severe wrinkling.

Possible Cause Number 3:

Cool-down without panel vacuum (and/or external pressure application) provides insufficient restraint of the layup to resist wrinkling.

To investigate this problem four (4) 36 in. x 18 in. laminates were laid up for prebleed

a) two six-ply asymmetric without vacuum during cool-down, one with a fast and one with a standard bleed cycle

b) six-ply asymmetric with vacuum during cool-down

c) six-ply asymmetric with 100 psi autoclave pressure (and vacuum) during cool-down

For three of the panels all but autoclave pressure prebleed was accomplished simultaneously to ensure essentially identical time/temperature relationships. Autoclave pressure cool-down was accomplished separately using as nearly identical a time/temperature cycle as could be achieved practicably.

Possible Cause Number 4:

Rapid cool-down tends to limit relaxation of thermal stresses since there is less time available for redistribution of resin due to increasingly viscous resin medium.

To investigate this problem three (3) 36 in. x 18 in. laminates were laid up for simultaneous prebleed:

a) six-ply asymmetric with rapid cool-down under vacuum

b) six-ply asymmetric with slow cool-down under vacuum

c) six-ply asymmetric with slow cool-down under vacuum with breather ply

Rapid cool-down was expected to produce more wrinkling and slow cool-down less wrinkling. There was in fact no noticeable difference.

Possible Cause Number 5:

Within-ply gaps between adjacent tapes caused by misalignment and/or lateral displacement during layup produce fiber depleted channels that constitute mechanically weak areas with greater susceptibility for initiation of wrinkles parallel to fiber direction.

To investigate this problem two (2) six-ply asymmetric 36 in. x 18 in. laminates were laid up for simultaneous prebleed:

a) small overlaps to prevent gaps

b) wide gaps

The layups made with overlaps were expected to wrinkle least, the wide gaps to show the greatest degree of wrinkling. No noticeable difference occurred.

### 7.1.2 Results of Definitive Tests

The results of the definitive tests are shown in Table 7-2. These results indicate that the separator film (W4500) and to a lesser degree the

TABLE 7-2. DEFINITIVE TEST RESULTS

| Lay-Up \ Panel No. | 1a | 1b | 1c | 2a | 2b | 2c |
|---|---|---|---|---|---|---|
| Release | Std | Std | Std | A4000 | No Film | Std |
| Peel Ply | Std | Std | Std | Std | Std | Std |
| GR/EP - Preplied Stock | 6 Ply | 6 Ply | 6 Ply | 6 Ply | 6 Ply | 5 Ply |
| Release Film, Perforated | Std | None | Std | Std | Std | Std |
| Bleeder | Std | None | 181 Glass | Std | Std | Std |
| Vacuum | Std | Std | Std | Std | Std | Std |
| Bleed Cycle | Std | Std | Std | Std | Std | Std |
| Degree of Wrinkling | Severe | Slight | Slight | None | None | Severe |

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TABLE 7-2. DEFINITIVE TEST RESULTS (Continued)

| Lay-Up \ Panel No. | 3a | 3a* | 3b* | 3c* | 4a | 4b | 4c | 5a | 5b |
|---|---|---|---|---|---|---|---|---|---|
| Release | Std | Std | Std | Std | Std | Std | Std | Std | Std |
| Peel Ply | Std | Std | Std | Std | Std | Std | Std | Std | Std |
| GR/EP - Preplied Stock | 6 Ply | 6 Ply | 6 Ply | 6 Ply | 6 Ply | 6 Ply | 6 Ply | 6 Ply-Lap | 6 Ply-Gap |
| Release Film, Perforated | Std | Std | Std | Std | Std | Std | Std | Std | Std |
| Bleeder | Std | Std | Std | Std | Std | Std | * | Std | Std |
| Vacuum | No Vacuum | No Vacuum | Std | 85psi + VAC. | Std | Std | Std | Std | Std |
| Bleed Cycle | Fast | Std | Std | $170^{o}F \pm 10^{o}F$ | Fast | Slow | Slow | Std | Std |
| Degree of Wrinkling | Severe | None | Severe | None | Mod. | Mod. | Mod. | Mod. | Mod. |

* Cel-Guard Film plus 181 Glass Fabric on Top of Bleeder

TABLE 7-3. SUBSTANTIATION TEST RESULTS

| Lay-Up \ Panel No. | A | B | C | D | E | F |
|---|---|---|---|---|---|---|
| Release Film | W4500 | A4000 | A4000 | A4000 | A4000 | A4000 |
| Peel Ply | Std | Std | Std | Std | Std | Std |
| GR/EP - Preplied Stock | 6 Ply | 6 Ply | 6 Ply | 6 Ply | 16 Ply | 32 Ply |
| Release Film, Perforated | Std | Std | Std | Std | Std | Std |
| Bleeder | Std | 181 Glass | Burlform | Std | Burlform | Burlform |
| Vacuum | Std | Std | Std | Std | Std | Std |
| Bleed Cycle | Std | Std | Std | Std | Std | Std |
| Wrinkling | Mod. | Mod. | Mod. | Mod. | None | None |

bleeder material (Airweave SS) may be the major elements in causing wrinkles to occur in prebled panels. As a continuation to the wrinkling investigation, additional tests were conducted to verify the above conclusions. In addition to verifying the use of a different separator film (A4000) and the use of different bleeder materials (181 glass cloth, Airweave, and Burlform), the effect of thickness was evaluated (6 ply, 16 ply and 32 ply). The results of these additional tests are shown in Table 7-3. As shown in this table, the initial indications that changing to the A4000 separator silm and a Burlform bleeder material would eliminate the wrinkling were proven to be incorrect. The last test included panels of 16 plies and 34 plies. Neither of these these panels wrinkled.

## 7.2 DETERMINING THE EFFECT OF WRINKLES ON CURED LAMINATES

A 100-inch 6 ply panel which was discarded because of workmanship (the layup technique appeared to entrap air that might aggravate the wrinkling) was completed through first prebleed. The panel was cut in half (lengthwise). The peel ply was removed from one half. An 0.020 aluminum caul sheet was placed over the bleeder and separator film on the other half. Both halves were prebled under a common vacuum bag. There was a vacuum leak during prebleed.

Following prebleed, wrinkles appeared in both panels. The wrinkles appeared generally in a random pattern, in random directions, and were of similar amplitude to those experienced on prior 100 inch layups. One deep (approx. .025" wrinkle extended completely across the narrow axis of both parts at the point of termination of an internal doubler ply, approximately 20 inches from the skin root end. All wrinkles in both panels were mapped for future reference. The panels are shown in Figure 7-3.

These panels were placed together and combined with a third prebled panel to form a 24 inch 100 inch 18-ply panel. This combined layup was cured in an autoclave. The physical appearance of this panel was good with no discernable wrinkles remaining in the previously mapped locations. The NDI C scan of this panel indicated an acceptable laminate.

The wrinkle map was overlayed on the cured part and the two locations with the worst combinations of wrinkles were marked. A panel was cut from each of these areas and sent to Lockheed-Georgia for evaluation. Lockheed-Georgia has a proprietary process for dissecting a graphite laminate ply by ply. The resulting ply by ply pieces were mounted and shipped back to Lockheed-California. There was no visual evidence of any wrinkling. It was thus concluded that the wrinkles straightened out as the tool expanded on heat up. The dissected plies are shown in Figures 7-4 thru 7-7.

Since it was evident that all wrinkles were removed during final cure and that laminates containing 16 or more plies were essentially free of wrinkles the ancillary test cover specimens were fabricated using the single stage prebleed of the skin and hats prior to final curve. With these larger laminates, up to 80 ins. long and 16 or more plies thick some minor wrinkling was evident prior to final cure. All wrinkles disappeared during the final

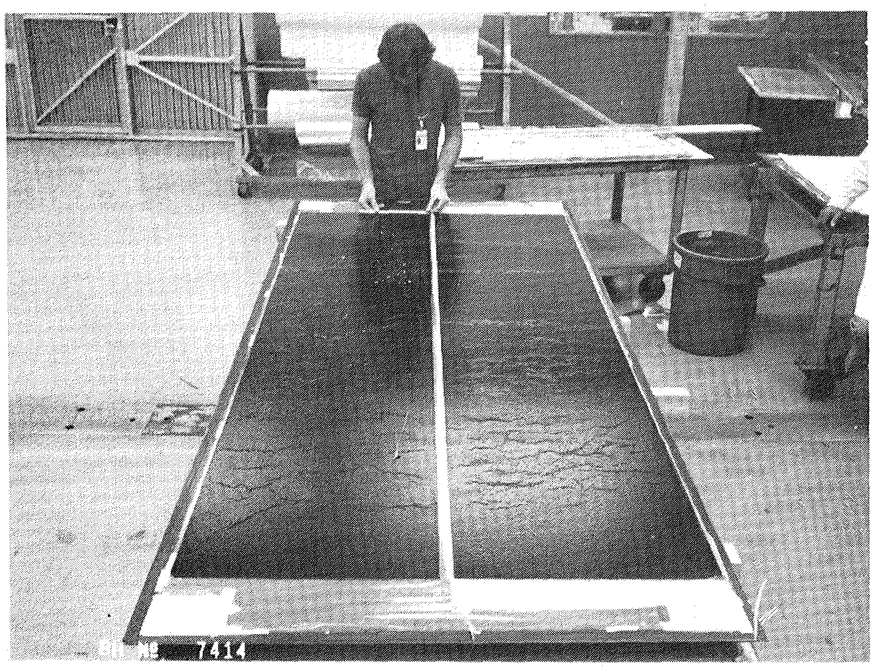


Figure 7-3. Wrinkled Prebled Panels Prior to Combining and Curing

cure and no ill-effects were noted either in NDI or in the destructive testing. The components fialed generally as predicted, at or close to the predicted loads.

During the period a parallel effort was in progress to develop a process for the low resin (36% by weight) material for inclusion later in the program.

## 7.3 CONCLUSIONS

The studies showed that wrinkling occurs when prebleeding thin panels in an oven. The wrinkles disappeared during final cure as far as visual inspection and C scan inspection was concerned. Ply by ply dissection also showed that even the most severe wrinkles straightened out and disappeared. Tested panels fabricated using wrinkled prebled stock behaved generally as predicted thus further indicating no influence of the precured wrinkling.

The studies also demonstrated that prebleeding can be an expensive step and helped lead to the decision to convert to the lower resin content prepreg when it became available.

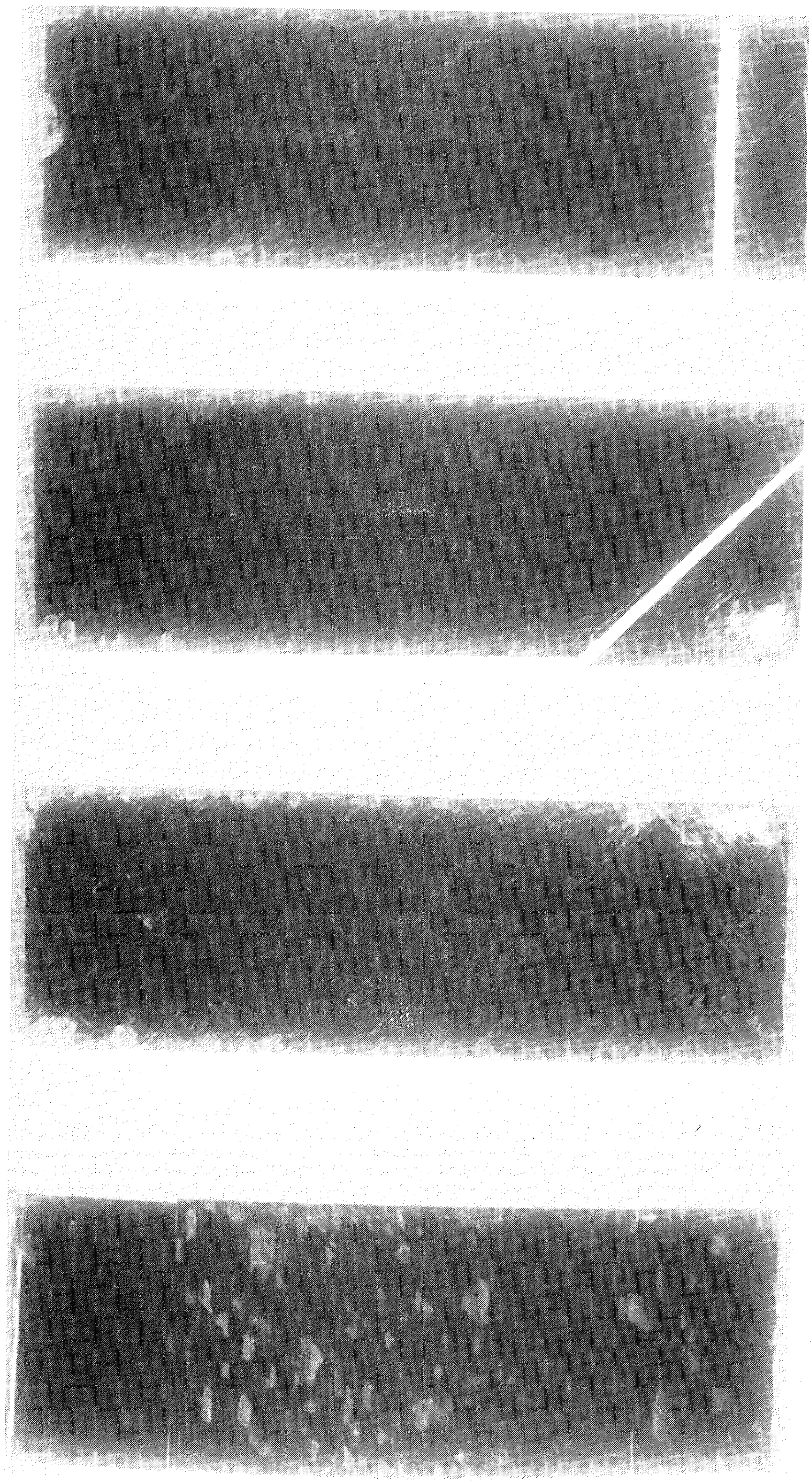

Figure 7-4. Dissected Panel Plies 1 Thru 4

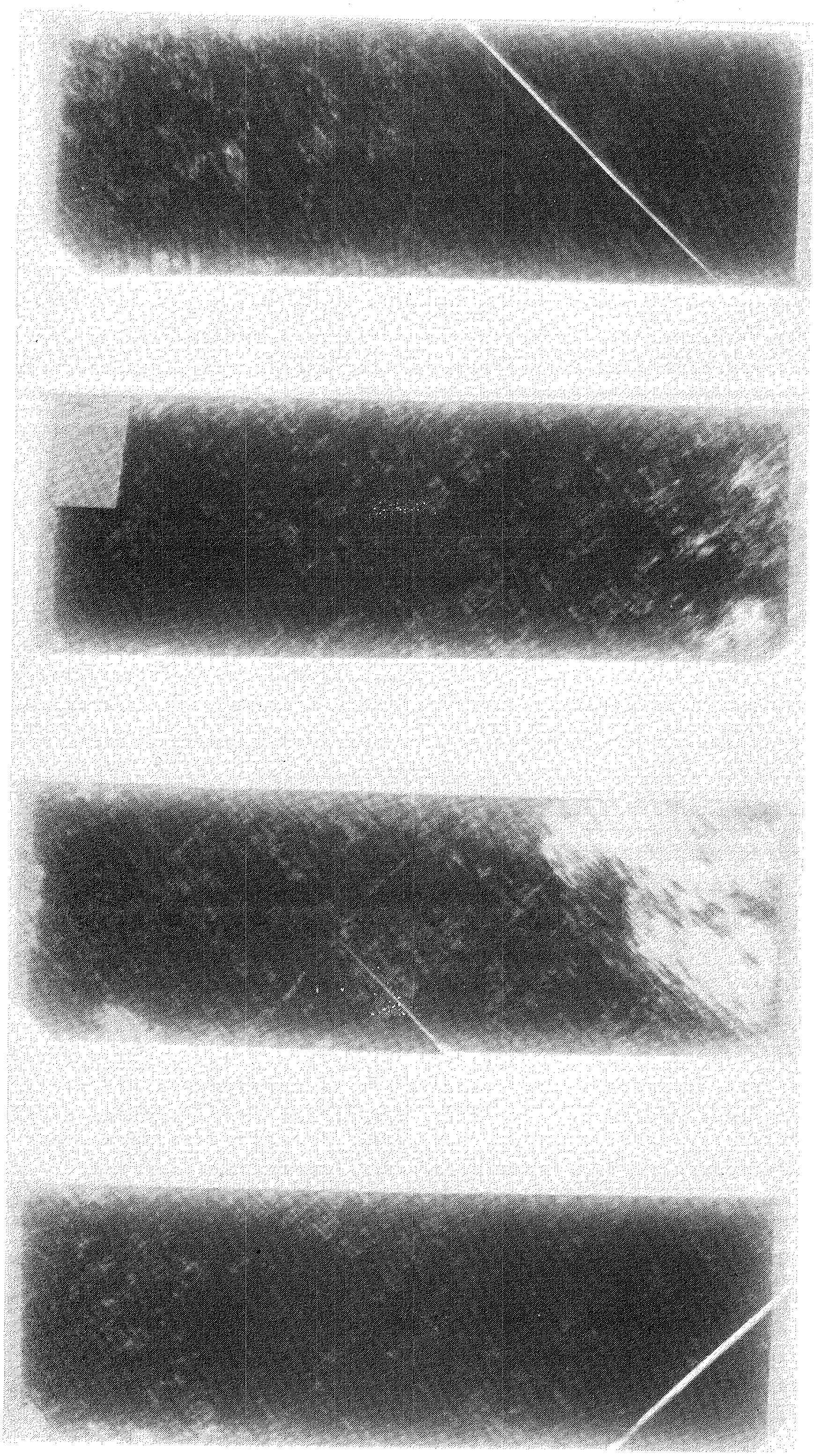

Figure 7-5. Dissected Panel Plies 5 Thru 8

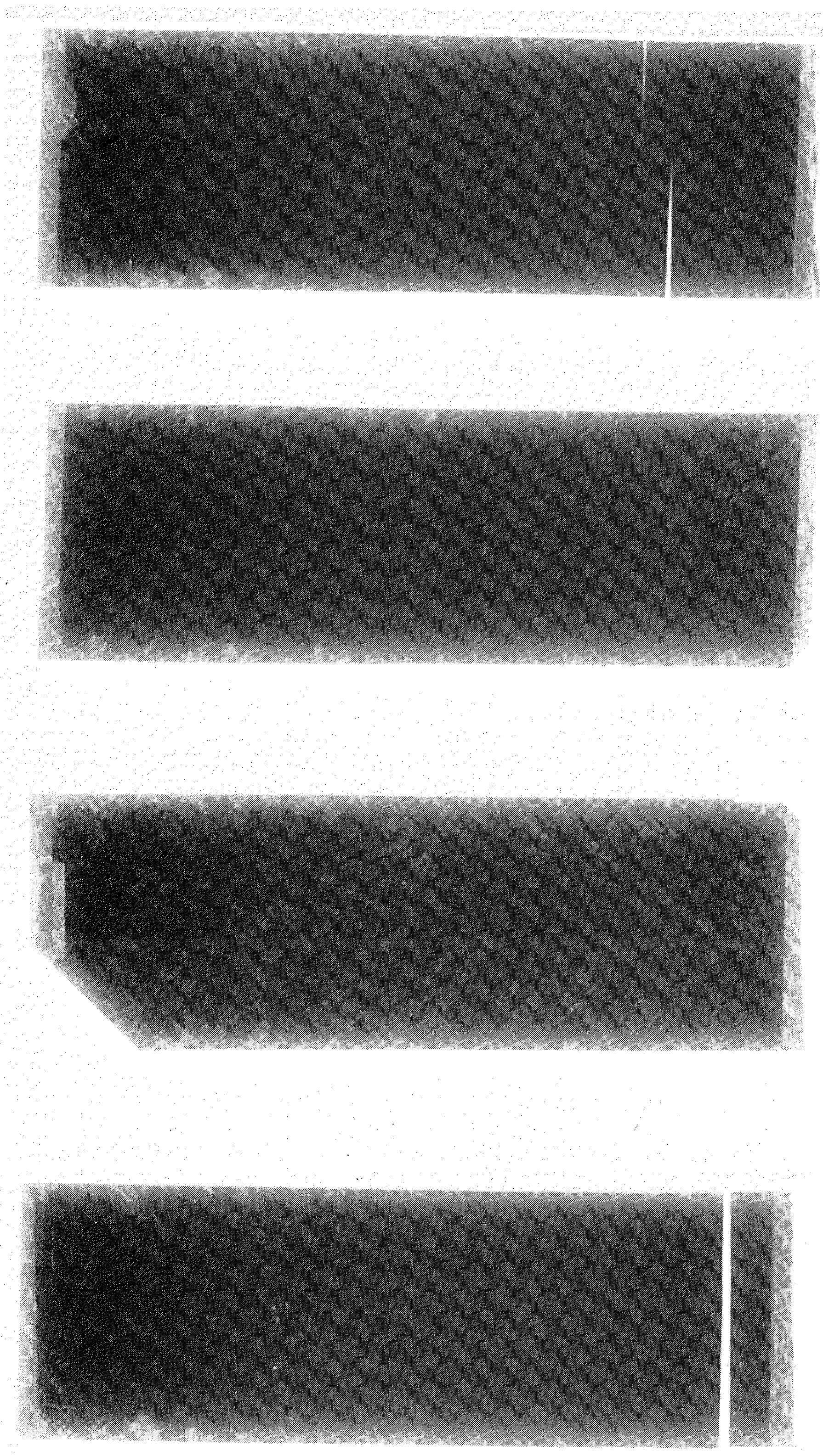

Figure 7-6. Dissected Panel Plies 9 Thru 12

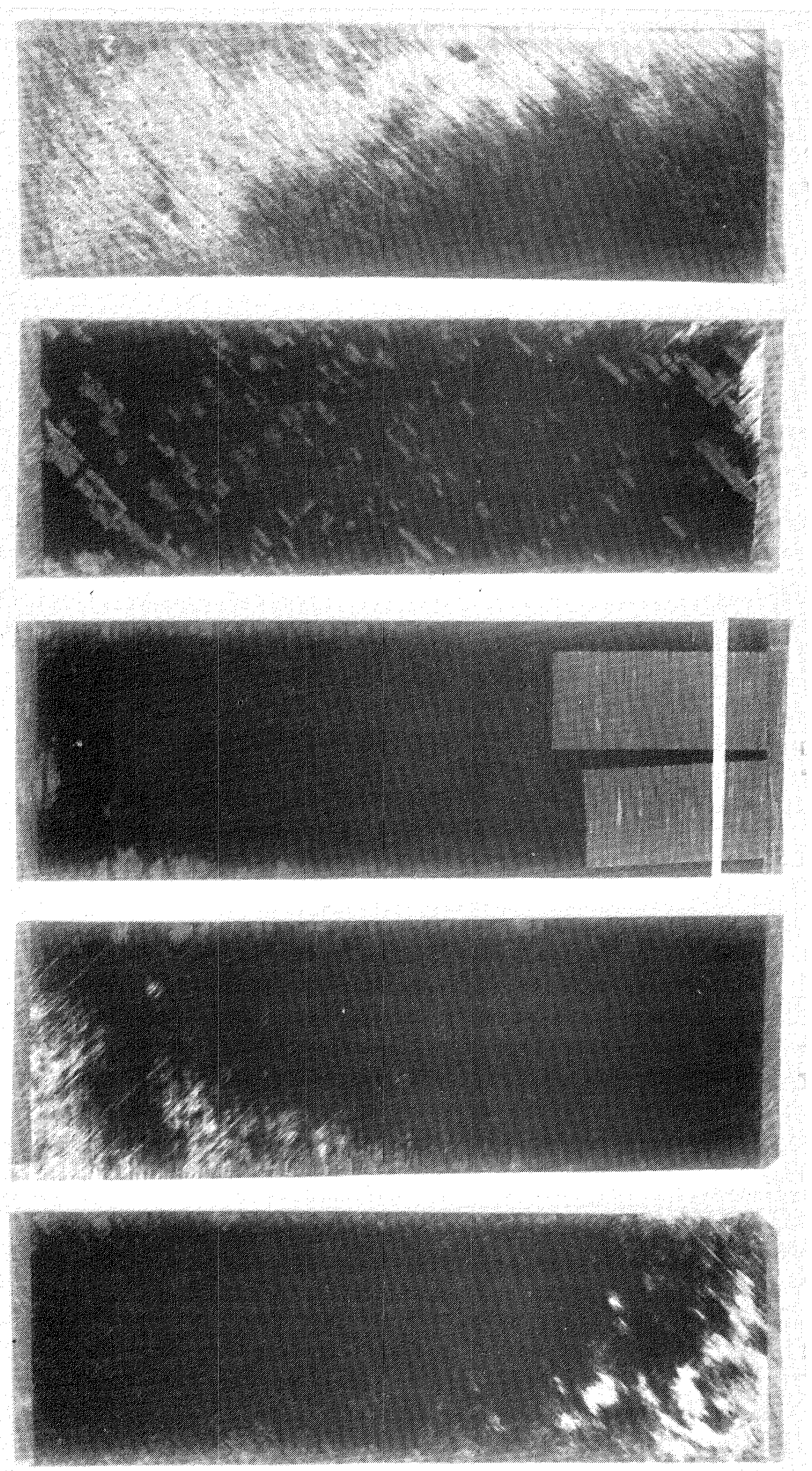

Figure 7-7. Dissected Panel Plies 13 Thru 17



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